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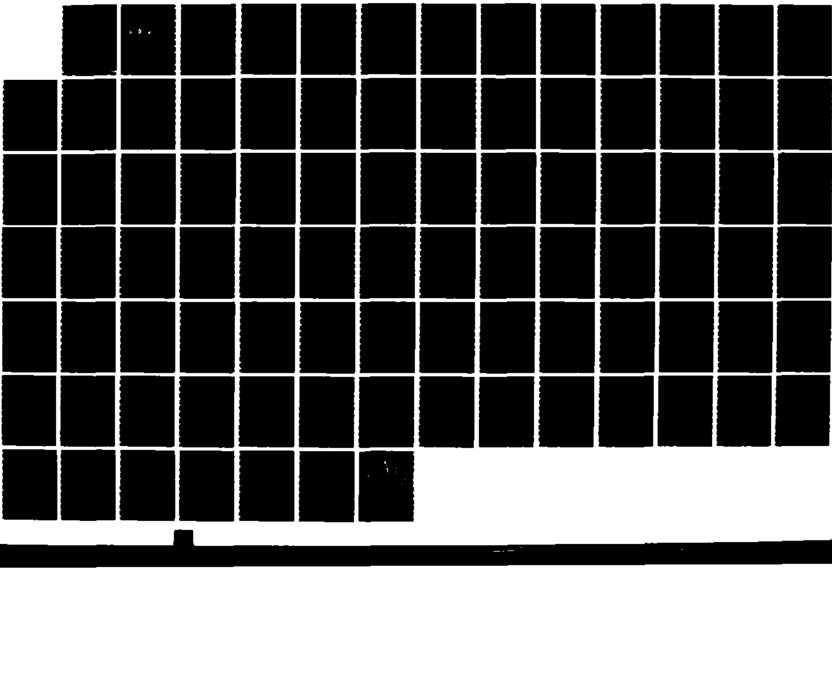

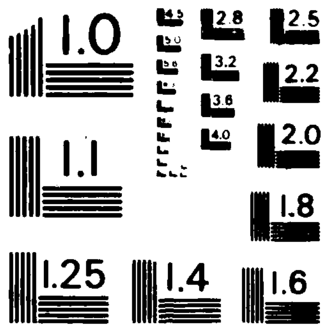

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NAVAL POSTGRADUATE SCHOOL

Monterey, California

THESIS

THE USE OF A COMPUTER
TO OBTAIN FLIGHT MANUAL DATA

by

Chang Whan Oh

December 1986

Thesis Advisor: Donald M. Layton

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SECURITY CLASSIFICATION OF THIS PAGE

AD-A176741

REPORT DOCUMENTATION PAGE

1a REPORT SECURITY CLASSIFICATION Unclassified	1b RESTRICTIVE MARKINGS
2a SECURITY CLASSIFICATION AUTHORITY	3 DISTRIBUTION/AVAILABILITY OF REPORT Approved for public release; distribution is unlimited
2b DECLASSIFICATION/DOWNGRADING SCHEDULE	
4 PERFORMING ORGANIZATION REPORT NUMBER(S)	5 MONITORING ORGANIZATION REPORT NUMBER(S)
6a NAME OF PERFORMING ORGANIZATION Naval Postgraduate School — 6b OFFICE SYMBOL (If applicable) Code 67	7a NAME OF MONITORING ORGANIZATION Naval Postgraduate School
6c ADDRESS (City, State, and ZIP Code) Monterey, California 93943-5000	7b ADDRESS (City, State, and ZIP Code) Monterey, California 93943-5000
8a NAME OF FUNDING/SPONSORING ORGANIZATION — 8b OFFICE SYMBOL (If applicable)	9 PROCUREMENT INSTRUMENT IDENTIFICATION NUMBER
8c ADDRESS (City, State, and ZIP Code)	10 SOURCE OF FUNDING NUMBERS PROGRAM ELEMENT NO — PROJECT NO — TASK NO — WORK UNIT ACCESSION NO

11 TITLE (Include Security Classification)

THE USE OF A COMPUTER TO OBTAIN FLIGHT MANUAL DATA

12 PERSONAL AUTHOR(S)

Oh, Chang W.

13a TYPE OF REPORT	13b TIME COVERED	14 DATE OF REPORT (Year, Month, Day)	15 PAGE COUNT
Master's Thesis	FROM ______ TO ______	1986 December	88

16 SUPPLEMENTARY NOTATION

17 COSATI CODES			18 SUBJECT TERMS (Continue on reverse if necessary and identify by block number)
FIELD	GROUP	SUB-GROUP	NATOPS FLIGHT MANUAL
			DRAG INDEX

19 ABSTRACT (Continue on reverse if necessary and identify by block number)

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A computer program for personal computer or hand-held calculator is developed to compute the desired fuel flow by modifing the equations for an 'IDEAL' airplane.

20 DISTRIBUTION/AVAILABILITY OF ABSTRACT ☒ UNCLASSIFIED/UNLIMITED ☐ SAME AS RPT ☐ DTIC USERS	21 ABSTRACT SECURITY CLASSIFICATION Unclassified
22a NAME OF RESPONSIBLE INDIVIDUAL Prof. Donald M. Layton	22b TELEPHONE (Include Area Code) (408) 646-2997 — 22c OFFICE SYMBOL Code 67Ln

DD FORM 1473, 84 MAR — 83 APR edition may be used until exhausted. All other editions are obsolete.

SECURITY CLASSIFICATION OF THIS PAGE

The Use of a Computer to obtain Flight Manual Data

by

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Submitted in partial fulfillment of the requirements for the degree of

MASTER OF SCIENCE IN AERONAUTICAL ENGINEERING

from the

NAVAL POSTGRADUATE SCHOOL
December 1986

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ABSTRACT

The one thing among many that must be prepared for every flight is the making of a flight plan. Pilots must use charts or graphs from the flight manual to compute the fuel flow that is essential to a flight plan. Since this requires many steps of interpolation to compute the specific conditions that cannot be read directly from flight manual, it is time consuming and increases the probability of making a mistake. This problem obstructs the execution of various mission changes and continuous sorties.

A computer program for personal computer or hand-held calculator is developed to compute the desired fuel flow by modifing the equations for an 'IDEAL' airplane.

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ACKNOWLEDGEMENTS

I am heartily thankful to my God and wish to express my appreciation to the Korean Air Force for the opportunity to study. Sincere thanks to Professor Donald M. Layton for his support and guidance. Without his support my effort would never have been completed.

My appreciation is also extended to my wife, Ho Jung and many individuals who prayed for me.

I. INTRODUCTION

A. BACKGROUND

A pilot must be furnished with flight performance information for his aircraft that will permit him to operate the aircraft efficiently at different airspeeds, different altitudes, different values of gross weight and different external loadings.

This information is collated from a combination of theoretical determinations and flight test data and presented to the pilot in a flight handbook in either a tabular format or in a series of graphs, or in combinations of these methods.

B. FUEL FLOW RATE

Probably the most important parameter in optimal performance of an aircraft is the rate at which fuel is consumed. It is obvious that when the fuel is exhausted the aircraft will no longer fly, so the pilot is very interested in the rate at which the fuel is being consumed. But also of importance is the fact that as the fuel is burned the weight of the aircraft is decreased and the performance of the aircraft is changed. It is therefore quite important that the pilot be able to forecast the rate of fuel consumption for each portion of each flight.

Fuel flow rate for a turbo-jet aircraft is a direct function of the thrust required for the specified flight condition, and in equilibrium flight thrust required is equal to the total drag of the aircraft.

C. TOTAL DRAG

The total drag of an aircraft in equilibrium flight may be subdivided into induced drag (the drag associated with the generation of lift) and parasite drag (the drag associated with the resistance of the fuselage to motion through the air). The induced drag is a function of the altitude of the aircraft (air density) and the weight of the aircraft, and the parasite drag is also a function of the altitude as well as being a function of the profile drag of the aircraft. The aircraft drag, which varies with the configuration and external loading (weapons, fuel tanks, et cetera) is usually expressed in terms of a Drag Index (DI), with the zero Drag Index usually being a measure of the drag of the aircraft in the 'clean' configuration.

D. THRUST

Although it is the thrust of the engines that pr[illegible] the p[illegible] turbo-jet aircraft, as long as there is sufficient thrust available [illegible] performance, the pilot is more interested for the flight planning in the fu[illegible]

E. FLIGHT PLANNING

To plan a flight, the pilot must determine what will be the loading [illegible] (gross weight) and what effect this loading will have on the drag (Drag I[illegible] pilot then enters the flight handbook to determine the fuel flow rate for the a[illegible] and altitude at which he desires to fly.

If the gross weight, Drag Index and airspeed are exactly the same as values [illegible] in the handbook tables or exactly the same as the valued on the graphical lines, the determination of the fuel flow rate is simply a matter of reading (correctly) the values from the information present in the handbook.

If, however, any of the input parameters are not identical to the handbook data points, the pilot must interpolate between the values presented. If weight, Drag Index, airspeed and altitude are all off-presentation points, the pilot may be forced to do several cross interpolations. This may even involve interpolations between values listed on different pages. Not only is this a time consuming and difficult task, but the possibility for error is expanded with each interpolation.

II. APPROACH TO THE PROBLEM

A. STATEMENT OF THE PROBLEM

A method was sought that would permit the pilot to rapidly obtain the fuel flow rate for various flight combinations of gross weight, Drag Index and airspeed. And, regardless of the ease of access to the information and completeness of the data obtained, this method must provide fuel flow rate at the same degree of accuracy as the flight handbook.

In addition, it was planned that the method of data retrieval should be as simple as possible with an ultimate goal of using a hand-held computer for the solutions.

B. BASE LINE APPROACH

The standard method of using the flight handbook is for the pilot to search and follow several lines and tables of the handbook to determine the required fuel flow rate. This requires that altitude, gross weight and Drag Index must be considered, but this method is too long, takes too much effort and is open to errors.

The procedure can be simplified by computerizing the flight manual. Most attempts at this method have involved the development of curve-fitting routines that will supply numerical modeling of the handbook data. Although the end use of these equations is quite rapid, the development of these routines is a long and arduous task. Inasmuch as each equation usually fits only a small part of the data, the computer programs for such a method are usually quite lengthy.

The approach undertaken in this project was the use of the equations for an 'IDEAL' aircraft to develop simplified equations that can be used in a short computer program.

Three steps were undertaken in the development of these equations:

a) The fuel flow rate data in the handbook is converted to thrust required.

b) The thrust required equations are adjusted so that they represent the actual aircraft.

c) The corrected thrust required equations are then reconverted to flow rate as the program output.

C. DETAILED METHODS

1. Concept of K_1 and K_2

Determinations are made for equilibrium, unaccelerated conditions where drag equals thrust.

The basic drag equations for an ideal aircraft are used [Ref. 1:p. 160]

$$C_D = C_{D_o} + C_{D_i}$$

$$C_{D_i} = C_L^2 / \pi\, e\, AR$$

$$D = (1/2)\rho\, V^2\, S\, C_D$$

All of the principal items of flight performance involve steady state flight conditions and equilibrium of the airplane. For the airplane to remain in steady level flight, equilibrium must be obtained by a lift equal to the airplane weight. And the airplane drag equals the thrust required to maintain steady level flight.

The total drag of the airplane is the sum of the parasite and induced drag. Parasite drag is the sum of pressure and friction drag which is due to the basic configuration and, as defined, is independent of lift. The equation of parasite drag can be expressed as a function of squared velocity.

$$D_o = (1/2) C_{D_o}\, \rho\, S\, V^2$$

$$\text{Let } K_1 = (1/2) C_{D_o}\, \rho\, S$$

$$D_o = K_1\, V^2$$

Induced drag is the undesirable but unavoidable consequence of the development of lift.

In the airfoil, the local lift vector is aligned perpendicular the the local relative wind and hence is inclined behind the vertical by the angle α . Consequently, there is a component of the local lift vector in the direction of V_∞ ; that is, there is a drag created by the presence of downwash.

This drag is defined as induced drag, denoted by D_i [Ref. 1:pp. 152-155]

$$D_i = (1/2)\rho\, S\, V^2\, C_{D_i}$$

$$C_{D_i} = C_L^2 / (\pi\, e\, AR)$$

$$D_i = (C_L)^2 / (2\pi\, e\, AR)\rho\, S\, V^2$$

$$= [(2W/\rho\ S\ V^2)^2 / (2\pi\ e\ AR)]\ \rho\ S\ V^2$$

$$= (4W^2\ \rho\ S\ V^2) / (\rho^2\ S^2\ V^4\ \pi\ e\ AR)$$

$$= [(2W^2) / (\rho\ S\ \pi\ e\ AR)](1/V^2)$$

If the whole item of right side except the squared velocity is changed to some constant, this equation can be expressed function of squared velocity.

Let $K_2 = (2W^2) / (\rho\ S\ \pi\ e\ AR)$

$$D_i = K_2(1/V^2)$$

While the parasite drag predominates at high speed, induced drag predominates at low speed. Figure 2.1 illustrates the variation with speed of the induced, parasite, and total drag for a specific airplane configuration in steady level flight.

The curve of drag or thrust required versus velocity shows the variation of induced, parasite, and total drag. Induced drag predominates at low speeds when the airplane is operated at maximum lift-drag ratio, $(L/D)_{max}$ the total drag is at a minimum and the induced and parasite drags are equal.

The effect of a change in weight on the thrust required is illustrated by Figure 2.2. The changes in the drag curves with variations of airplane weight, configuration, and altitude furnish insight for the variation of range, endurance, climb performance, etc., with these same items.

First, the primary effect of a weight change is a change in the induced drag at any given speed. Thus, the greatest changes in the curves of thrust and power required will take place in the range of low speed flight where the induced effects predominate. The changes in thrust and power required in the range of high speed flight are relatively slight because parasite effects predominate at high speed. The induced effects at high speed are relatively small and changes in these items produce a small effect on the total thrust required. [Ref. 2:p. 99]

Second, Figure 2.3 illustrates the effect on the curve of thrust required of a change in the equivalent parasite area f of the configuration. Since parasite drag predominates in the region of high flight speed, a change in f will produce the greatest change in thrust and power required at high speed. Since parasite drag is relatively

THRUST REQUIRED VS VELOCITY

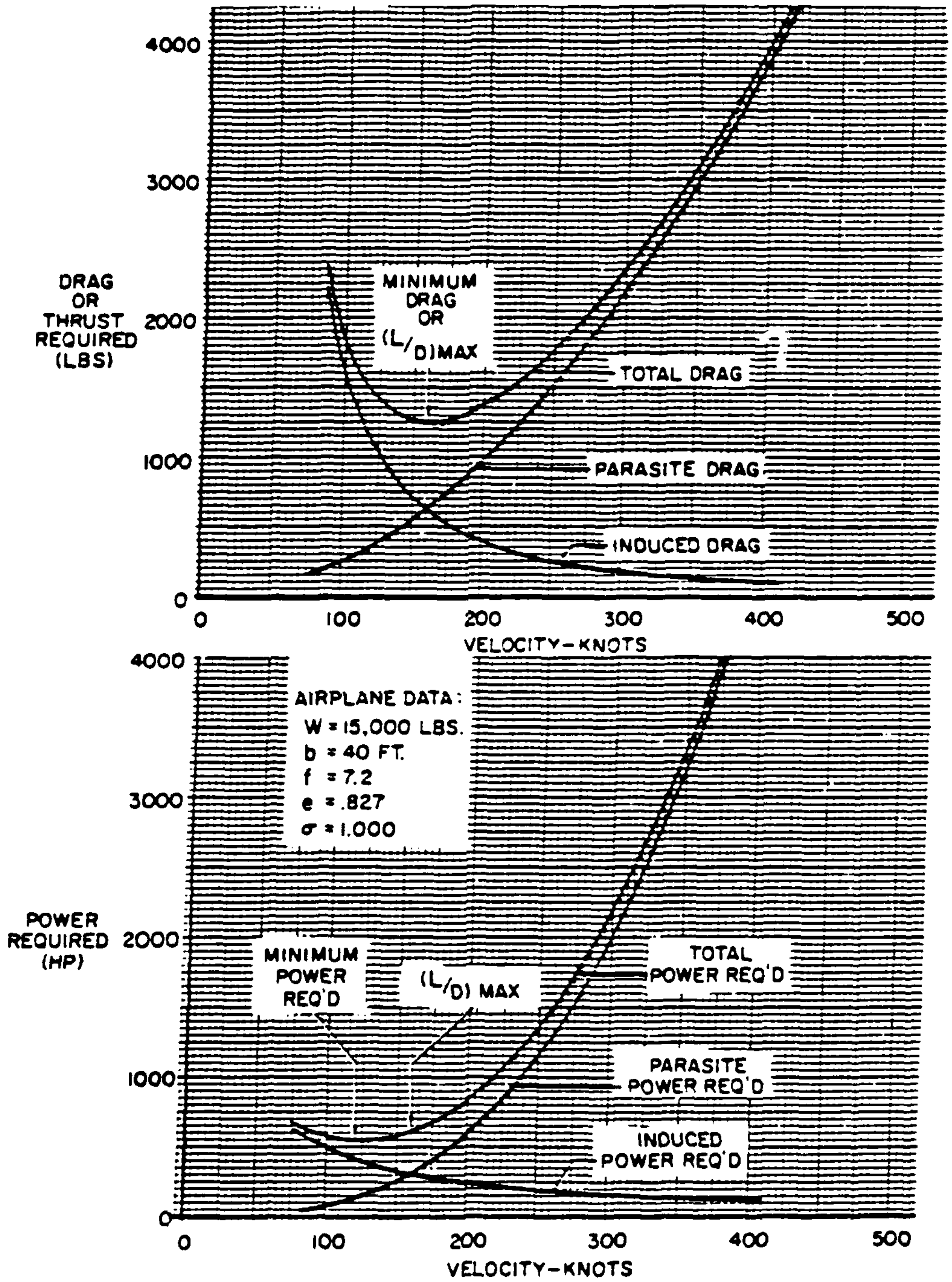

Figure 2.1 Airplane Thrust and Power Required.

EFFECT OF WEIGHT CHANGE

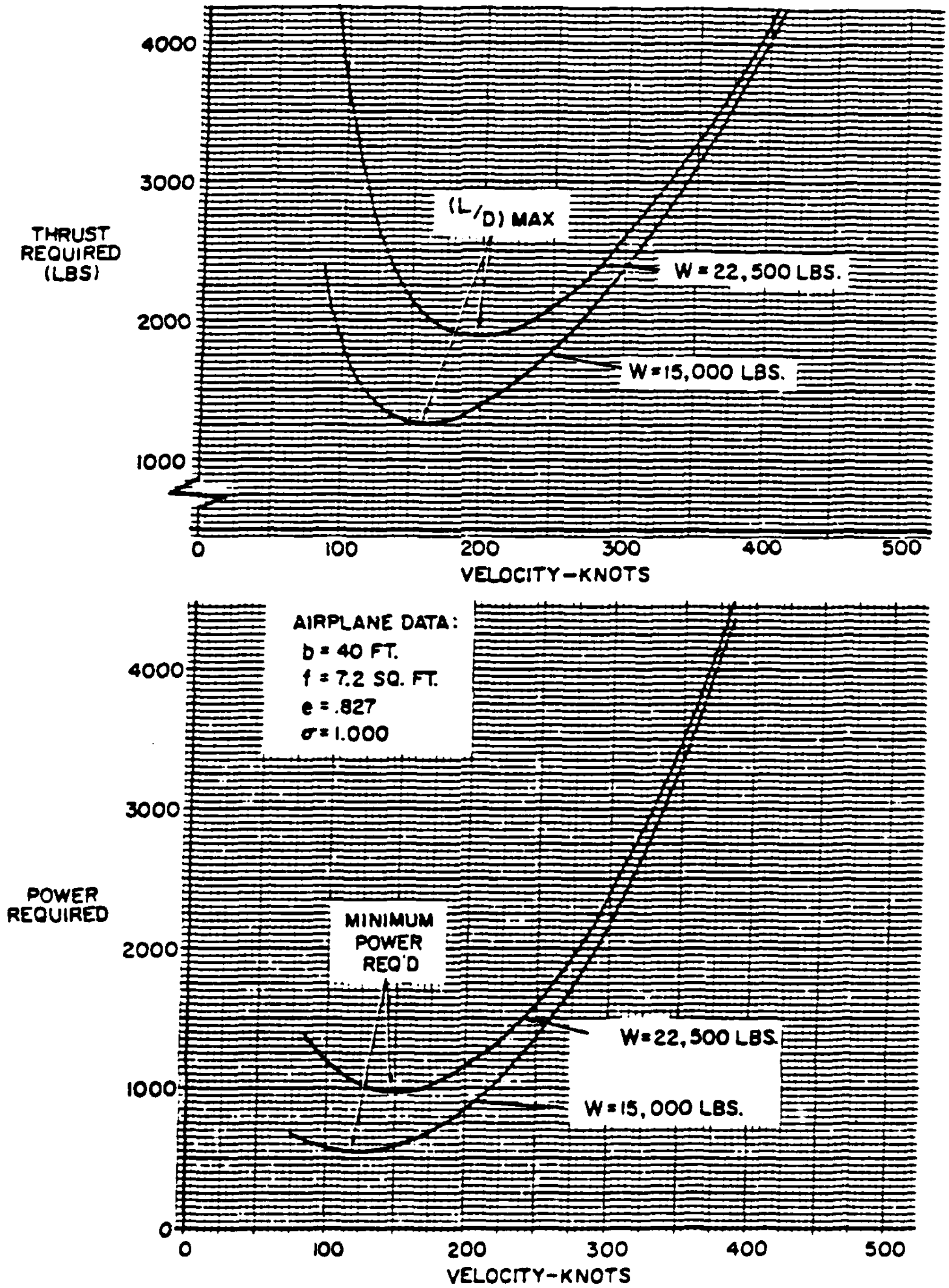

Figure 2.2 Effect of Weight on Thrust and Power Required.

small in the region of low speed flight, a change in f will produce relatively small changes in thrust required at low speeds. The principal effect of a change in equivalent parasite area of the configuration is to change the parasite drag at any given airspeed. [Ref. 2:p. 101]

Third, a change in altitude can produce significant changes in the curves of thrust required. The effect of altitude on these curves provide a great part c` the explanation of the effect of altitude on range and endurance. [Ref. 2:p. 101]

Figure 2.4 illustrates the effect of a change in altitude in the curves of thurst required for a specific airplane configuration and gross weight. As long as compressibility effects are negligible, the principal effect of increased altitude on the curve of thrust required is that specific aerodynamic conditions occur at higher true airspeeds.

As being investigated above, the total drag, same as thrust required, is the function of squared velocity with different constant.

Total drag equation is

$$D = K_1 V^2 + K_2 / V^2$$

$$\text{where} \quad K_1 = C_{D_0} \rho S$$

$$K_2 = (2 W^2) / (\rho S \pi e AR)$$

The first step of this project is to determine the K_1 and K_2 values to obtain the thrust required. It can be seen that the K_1 is the function of altitude and drag index, K_2 is the function of gross weight and altitude. So if K_1 and K_2 values can be computed with any flight condition, the thrust required will be solved at any airspeed.

2. Determining the K_1

To determine the K_1, use the minimum drag concept.

At the minimum drag point, the airplane can be flown maximum endurance and the induced drag and parasite drags are the same.

The first step for determine the K_1 is to investigate the thrust equation. The thrust equation can be established with fuel flow for each airspeed and some factor C.

THRUST = FUEL FLOW x C

Since the fuel flow of the turbojet powered airplane is proportional to the airspeed, the thrust required is proportional to the fuel flow.

EFFECT OF PARASITE AREA CHANGE

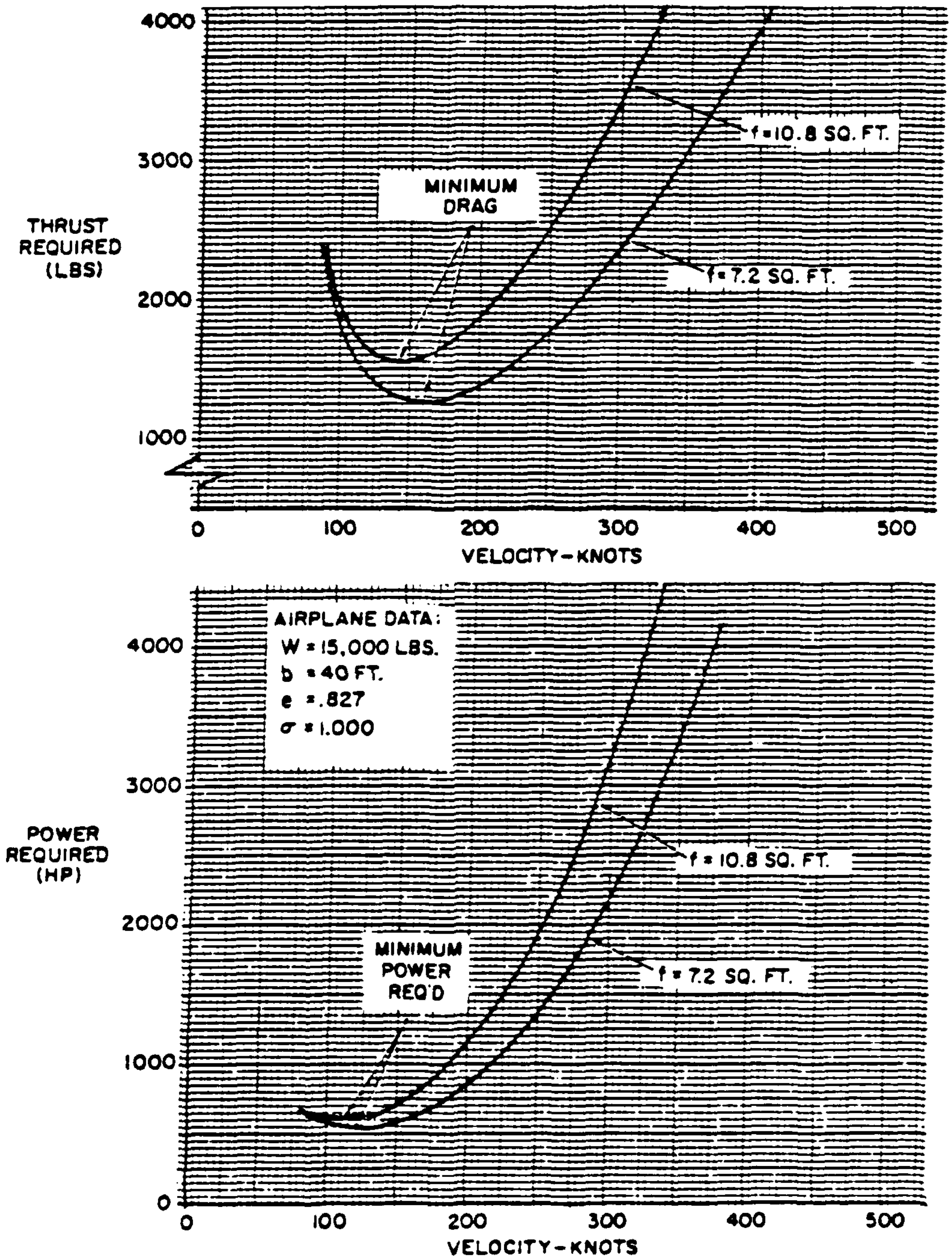

Figure 2.3 Effect of Equivalent Parasite Area,f, Thrust and Power Required.

EFFECT OF ALTITUDE CHANGE

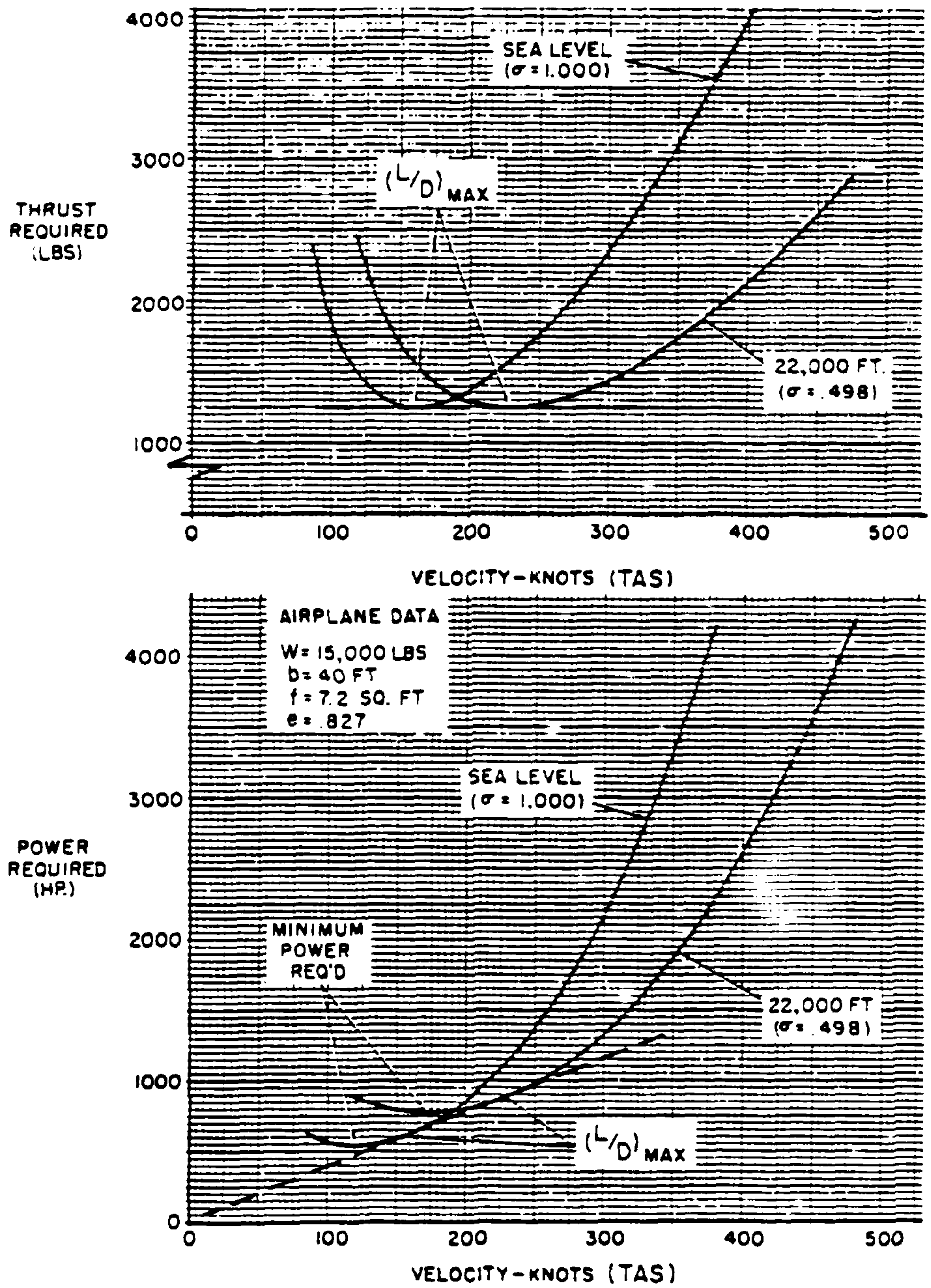

Figure 2.4 Effect of Altitude on Thrust and Power Required.

The factor C is the value of the military power devided by fuel flow at military thrust.

C = MILITARY THRUST / FUEL FLOW (at military thrust)

The military thrust and fuel flow information for any condition can be obtained from NATOPS FLIGHT MANUAL. With this factor C, the thrust required for each speed can be computed by fuel flow,came from NATOPS FLIGHT MANUAL, multiplies factor C. Table 1 and Figures 2.5 and 2.6 show the relationship and the result between thrust and velocity.

TABLE 1

THRUST REQUIRED AND FUEL FLOW VERSUS VELOCITY

VELOCITY(KTAS)	FUEL FLOW(LBS/H)	THRUST REQUIRED(LBS)
360	7,725	6,558
400	9,014	7,652
440	10,623	9,018
480	12,462	10,579
520	14,652	12,438
560	17,377	14,752
600	21,393	18,160
MIL	25,680	21,800

Example) military thrust ; 21800 lb

fuel flow (at mil) ; 25680 lbs/hr

Factor C = 21800/25680

= 0.84891

Thus

Thrust required = 0.84891 x Fuel Flow

Maximum endurance speed at specific gross weight, altitude, and drag index can be figured out with the graph of maximum endurance mach number in NATOPS MANUAL Figure 2.7. This true mach number needs to be transfered to KTAS to substitute into the drag equation.

Next step is to compute the minimum drag.

Minimum drag can be computed from the fuel flow at maximum endurance multiplied by the factor C. The maximum endurance fuel flow can be obtained from the NATOPS MANUAL Figure 2.8.

And so the minimum drag is the twice of the parasite drag.

$$D_{total} = 2D_i = 2D_o = T_{min}$$

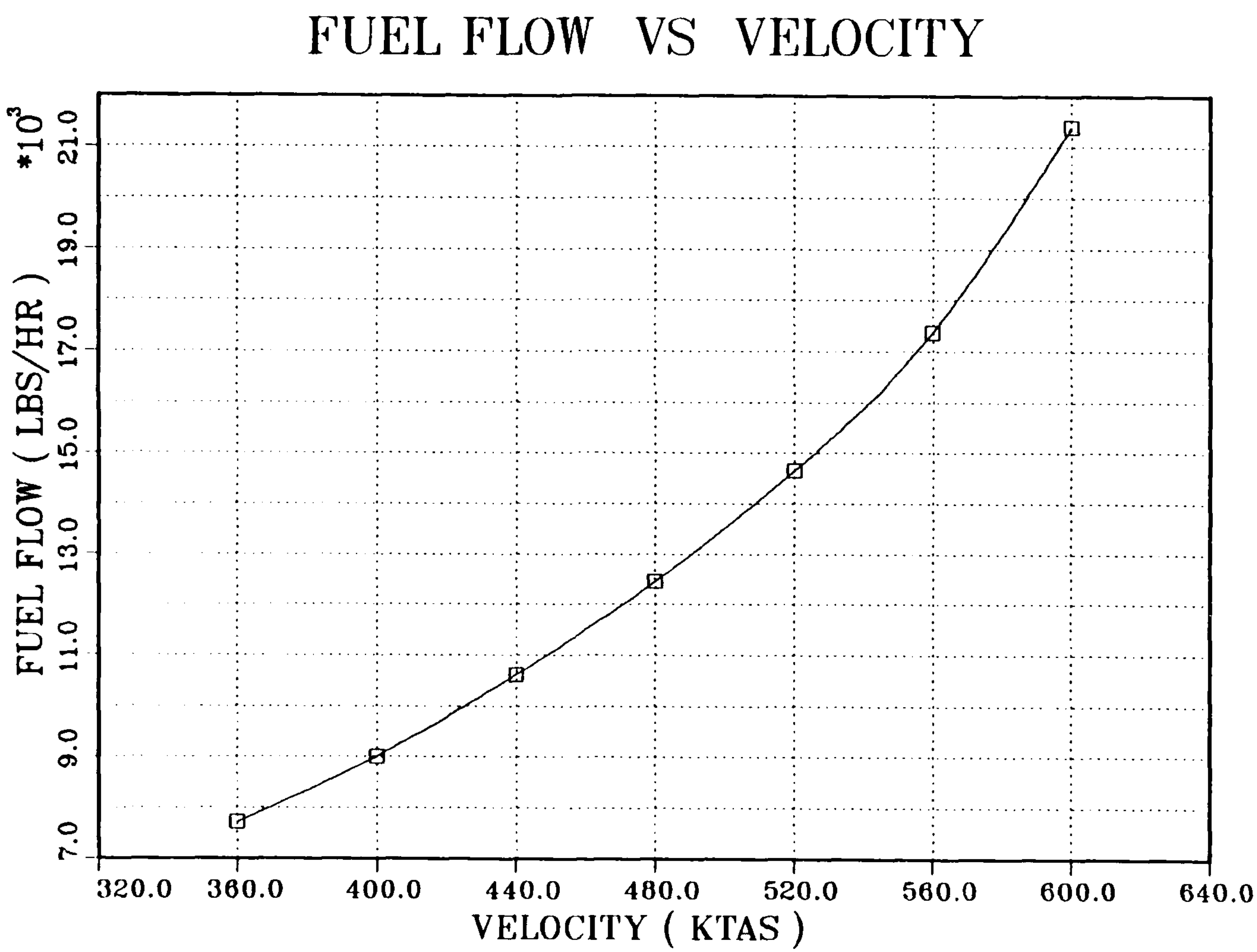

Figure 2.5 Fuel Flow Required for Velocity.

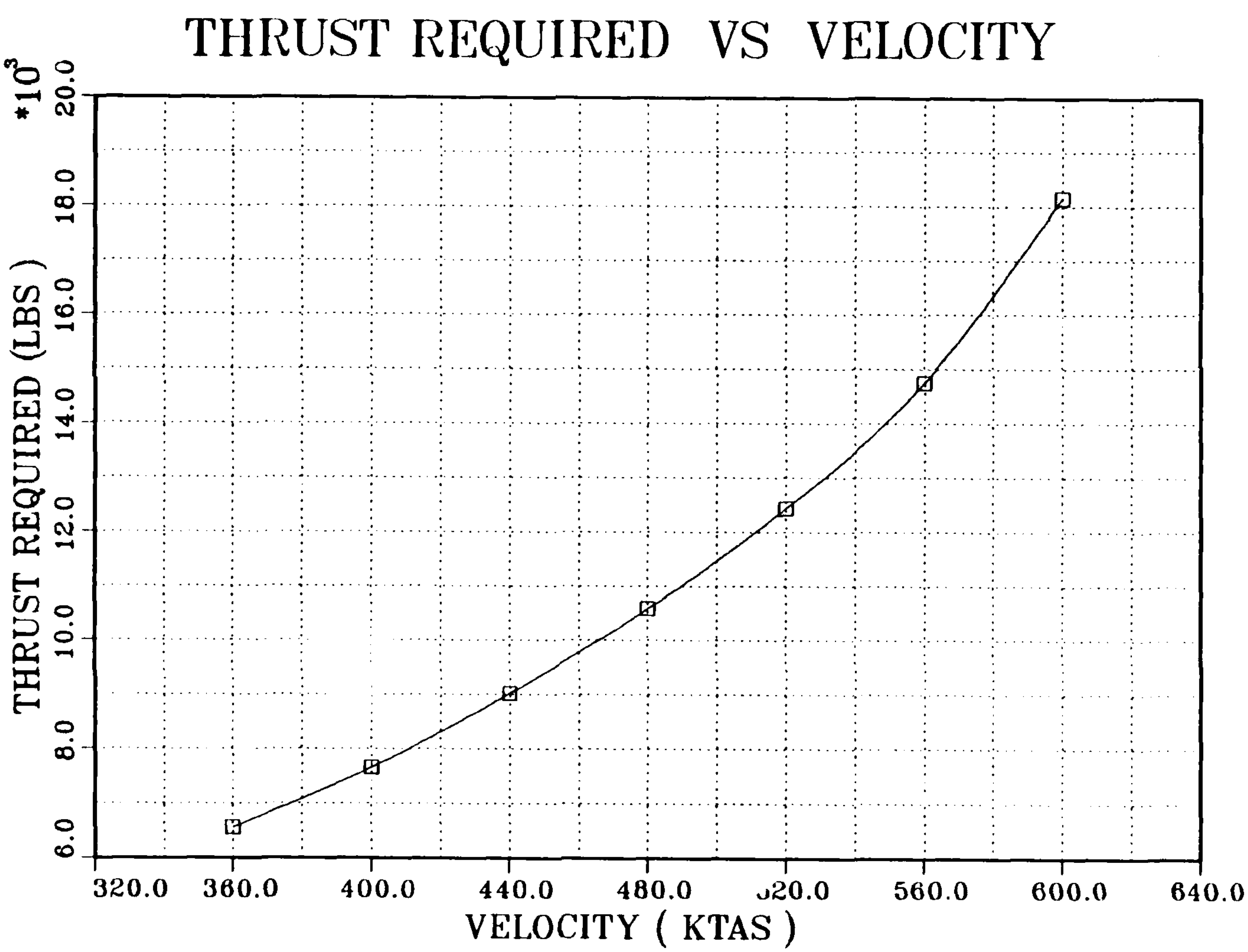

Figure 2.6 Thrust Required for Velocity.

MAX ENDURANCE MACH NUMBER

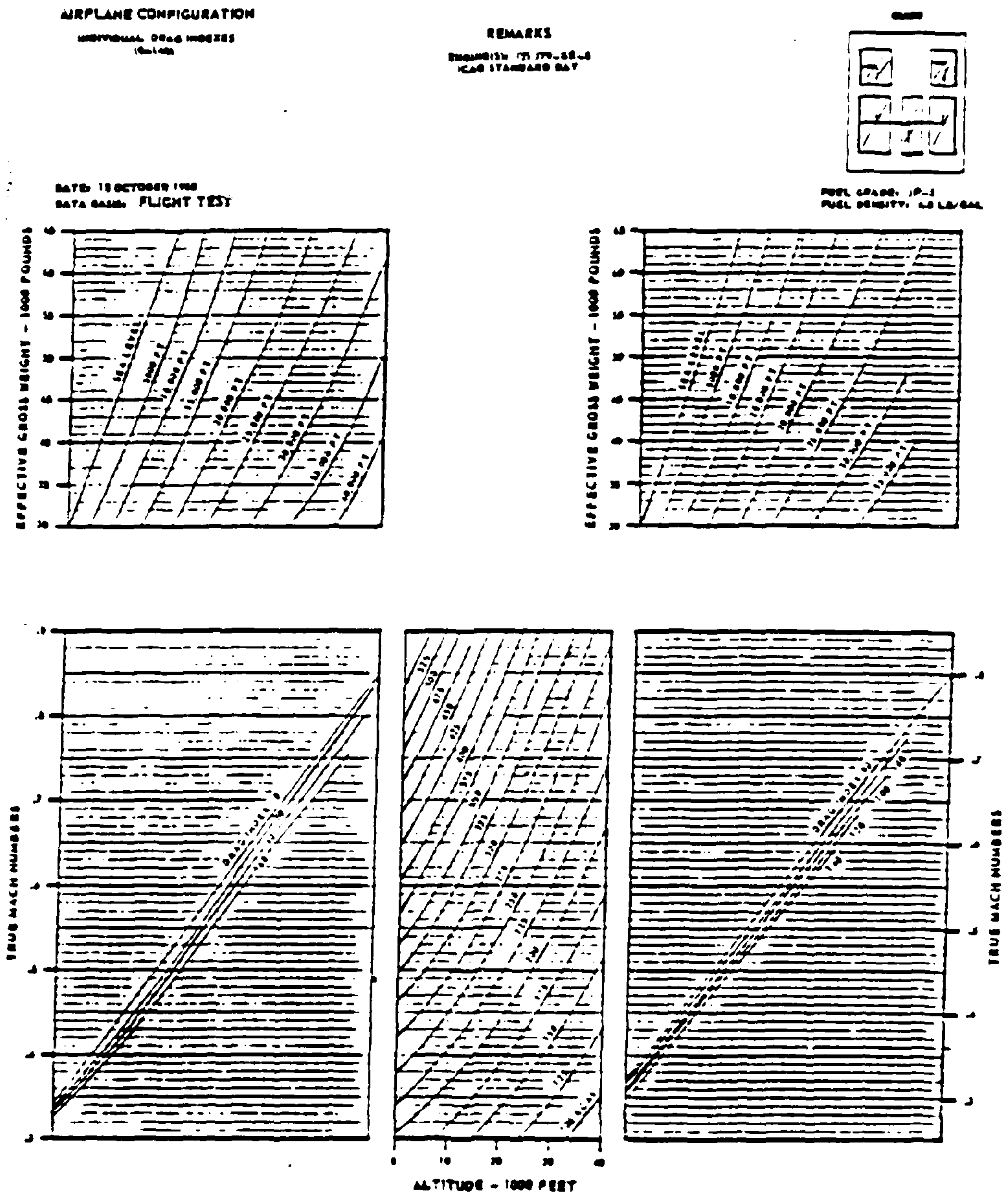

Figure 2.7 Maximum Endurance Mach Number.

MAX ENDURANCE FUEL FLOW

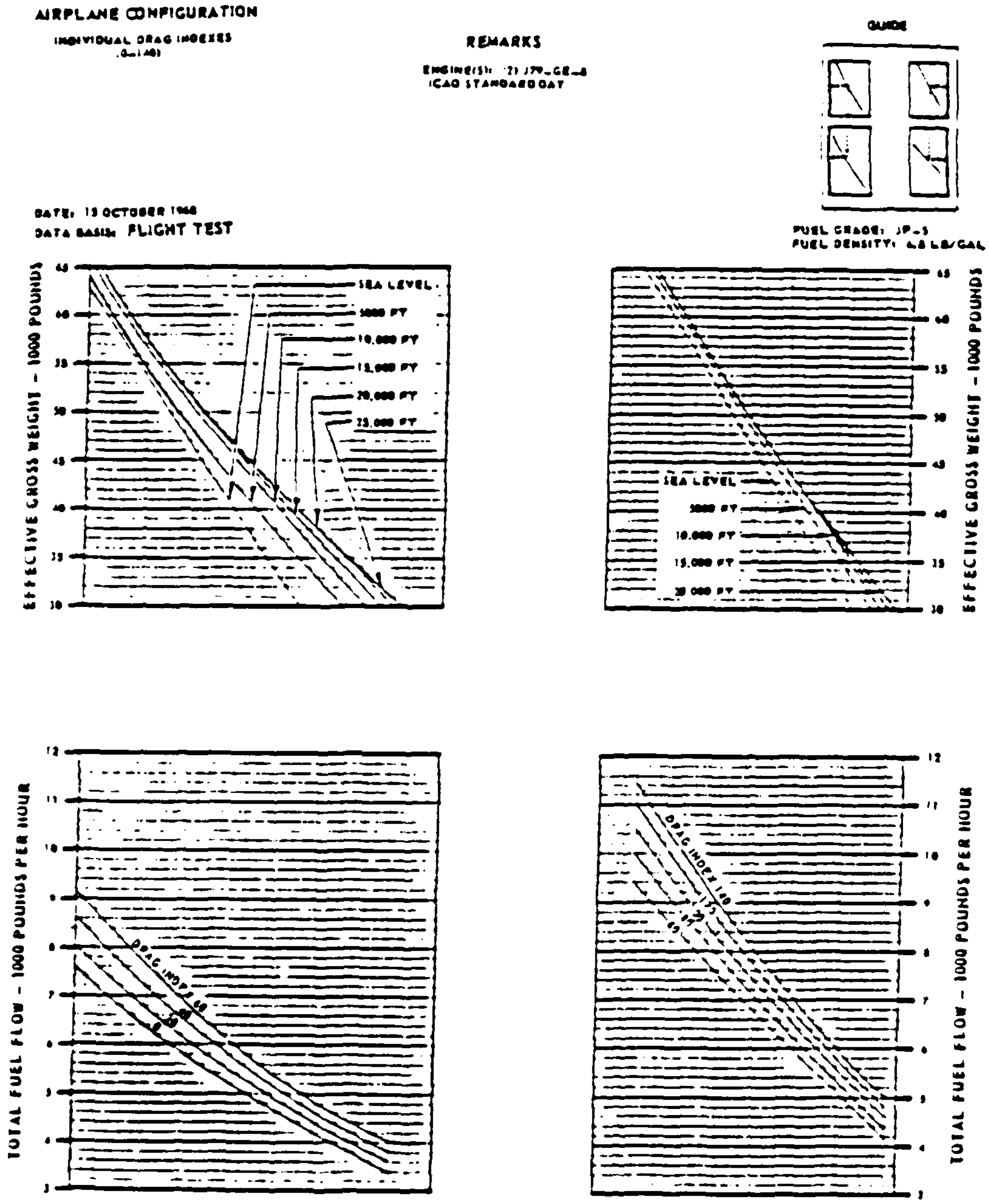

Figure 2.8 Maximum Endurance Fuel Flow.

Thus the drag equation can be expressed.

$$T_{min} / 2 = D_o = (1/2)\, C_{D_o}\, \rho\, S\, V^2$$

where ρ = density at given altiude

V = velocity at minimum drag

S = wing area of specific airplane

C_{D_o} can be obtained from the above equation, because all values except C_{D_o} are known, and the drag equation is a function of squared velocity with the known C_{D_o}.

The total value of right side except V^2 is the desired constant K_1.

3. **Determining the K_2**

The procedure to compute K_2 is just similiar to that for computing K_1. It is convenient to use the same minimum drag concept. At the minimum drag point, the minimum thrust required is twice the induced drag amount.

Use the same maximum endurance velocity and minimum thrust required.

The induced drag equation is

$$D_i = T_{min} / 2$$

$$= (2W^2) / (\pi\, e\, AR\, \rho\, S)(1/V^2)$$

where w = Gross weight of specific airplane

s = Wing area of specific airplane

ρ = Air density at given altitude

v = Maximum endurance velocity

Since the minimum thrust required and all items of the right side are known value except $1/\pi$ e AR, the value of $1/\pi$ e AR can be solved easily. After obtained the $1/\pi$ e AR value, substitute this value into the above induced drag equation, and calculated the numerical value of the right side except the $1/V^2$. This numerical value is just the desired constant K_2 value.

But if the raw K_1 and K_2 values are used to compute the thrust required, there will exist a deviation of more than 10 percent at high speed. Therefore one needs to modify the K_1 and K_2 to fit the curve to the FLIGHT MANUAL data.

The fitting method used in this project is the trial and error method. The Table 2 and Figure 2.9 illustrates the difference between the modified and raw value of K_1, K_2.

TABLE 2

THRUST REQUIRED AND FUEL FLOW DEVIATION

ALTITUDE	0 FT	GROSS WEIGHT	40,000 LBS		DRAG INDEX	20
MILITARY	THRUST	21,800 LBS	THRUST	FUEL FLOW x 0.84891		
	FUEL FLOW	25,680 LBS/H				
AIR SPEED (KTS)	FUEL FLOW (LBS/H)	THRUST (LB) HAND BOOK	THRUST REQUIRED (CAL')		DEVIATION (%)	
			RAW K	MODIFIED K	RAW K	MODIFIED K
360	7,725	6,558	6,125	6,507	- 6.6	-0.8
400	9,014	7,652	7,083	7,804	- 7.4	1.9
440	10,623	9,018	8,220	9,275	- 8.8	2.8
480	12,462	10,579	9,518	10,912	-10.0	3.1
520	14,652	12,438	10,966	12,708	-11.8	2.2
560	17,377	14,752	12,558	14,754	-14.9	0.0
600	21,393	18,160	14,286	18,007	-21.3	-0.8
MAXIMUM ENDURANCE		FUEL FLOW	5,600 (LBS/H)			
		THRUST (MINIMUM)	5,600 x 0.84891 = 4,754 (LB)			
		AIR SPEED	248 (KTAS)			
RAW	K_1	0.0135	MODIFIED	K_1	0.01614	
	K_2	4.1755×10^8		K_2	2.0×10^8	

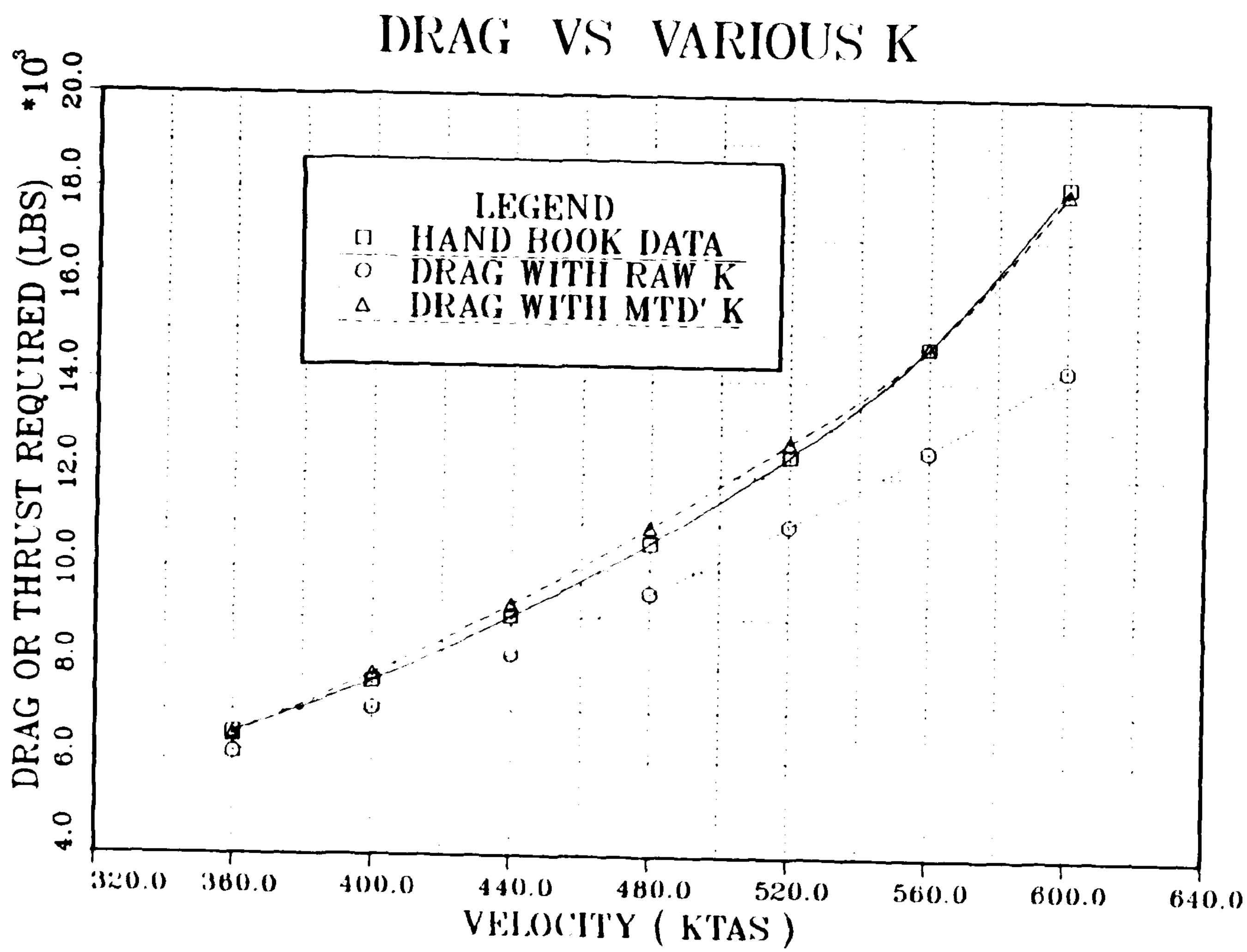

Figure 2.9 Thrust Required for Velocity with Various K1.

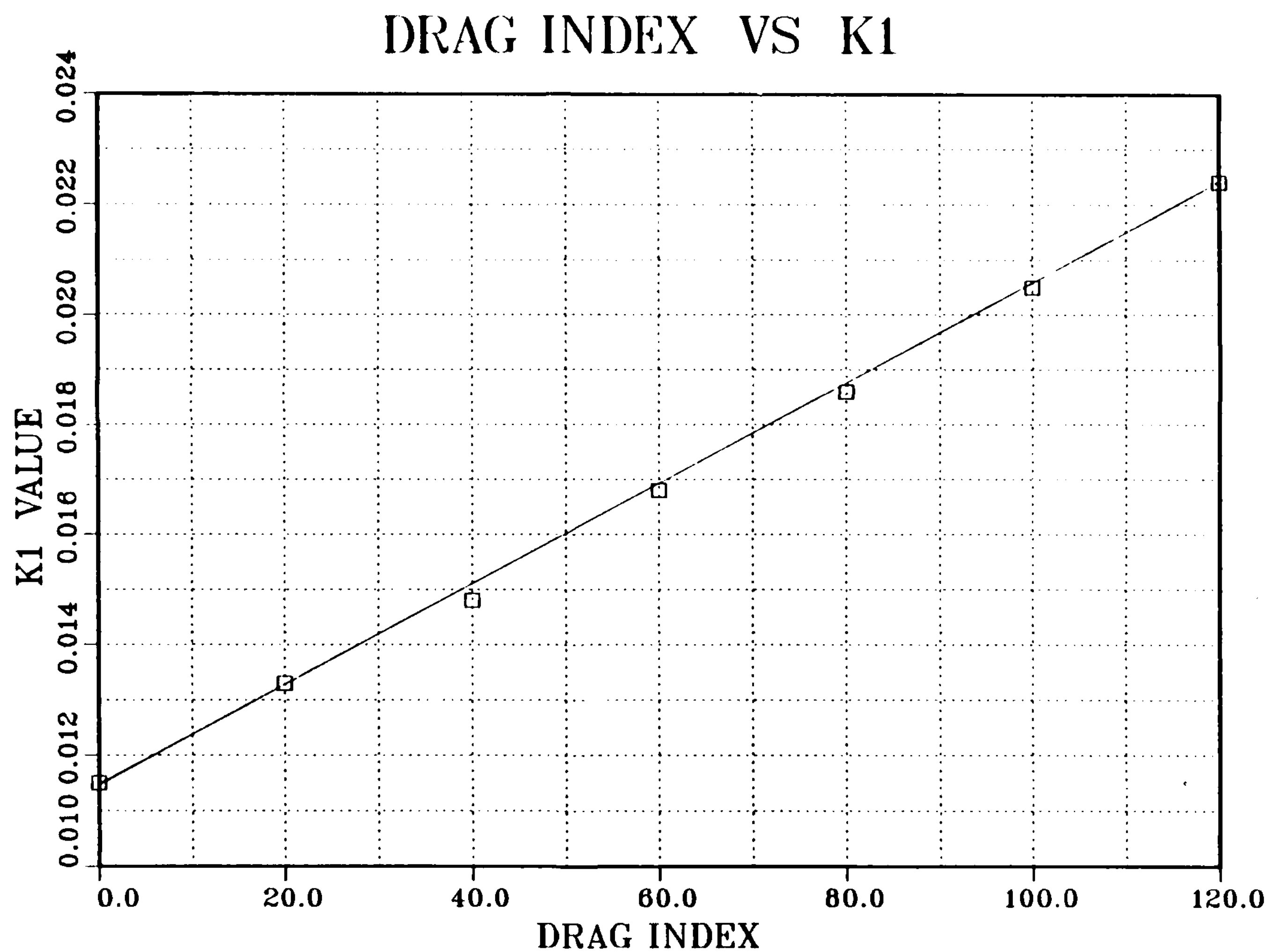

Figure 2.10 Drag Index versus K1.

4. **Relationship between K_1, K_2 and Variation Factors**

Next consideration is to make the equations of K_1 and K_2 functions of drag index, altitude and gross weight.

Since the K_1 is the function of drag index and air density, K_1 will be changed by different drag index and different altitude. Thus it is needed to develop a relationship of K_1 with the drag index and altitude.

To accomplish this, it is convenient to select some reference altitude and gross weight, because the relationship of K_1 and drag index is independent of altitude and gross weight. Each K_1 value for every drag index can be calculated with the previous procedure. After calculation of all of the K_1 values, a graph of K_1 versus drag index is plotted to find the relationship.

In this project, the altitude and gross weight are selected to standard sea level and 40,000 lbs. The final values and relationships are shown in Table 3 and Figure 2.10.

TABLE 3

DRAG INDEX VS K1

Drag Index	K_1
0	0.011534
20	0.013335
40	0.014807
60	0.016780
80	0.018480
100	0.020527
120	0.022361

Fortunately, as can be seen in the result of the graph, the relationship between K_1 and the drag index is linear. So the equation K_1 as a function of drag index is

$$K_1 = (0.0115 + 0.0001 \times \text{Drag Index})$$

In this equation, the most important thing is the slope of the graph, because the starting value of K_1 was modified to fit the published data.

The K_1 equation with modified starting value of K_1 is

$$K_1 = (0.014 + 0.0001 \times \text{Drag Index})$$

This equation was verified by calculating the drag at each speed. The results are shown in Tables 4, 5, 6.

TABLE 4

THRUST REQUIRED DEVIATION AT DI 20

ALTITUDE	0 FT	GROSS WEIGHT	40,000 LBS	D.I	20

MILITARY	THRUST	21,800 LB	THRUST	FUEL FLOW x 0.8489
	FUEL FLOW	25,680 LB/H		

AIR SPEED (KTS)	FUEL FLOW (LBS/H)	THRUST (LB) HAND BOOK	THRUST (LB) CALCULATION	DEVIATION (%)
360	7,725	6,558	6,456	-1.5
400	9,014	7,652	7,741	1.2
440	10,623	9,018	9,198	2.0
480	12,462	10,579	10,819	2.3
520	14,652	12,438	12,600	1.3
560	17,377	14,752	14,536	-1.5
600	21,393	18,160	16,625	-8.5

MAXIMUM ENDURANCE	FUEL FLOW	5,600	(LBS/H)
	THRUST (min)	4,754	(LB)
	AIR SPEED	248	(KTAS)

VARIATION OF K	K_1	0.014+0.0001x20 = 0.016
	K_2	2.0×10^8

From Tables 4, 5, 6, the following can be observed.

First, the thrust required, i.e., calculated drag, is almost same as published data at low drag index range, but the thrust required is much less than published data at high drag index range.

TABLE 5
THRUST REQUIRED DEVIATION AT DI 60

ALTITUDE	0 FT	GROSS WEIGHT	40,000 LBS	D.I	60
MILITARY	THRUST	21,800 LB	THRUST	FUEL FLOW x 0.8925	
	FUEL FLOW	24,425 LB/H			
AIR SPEED (KTS)	FUEL FLOW (LBS/H)	THRUST (LB) HAND BOOK	THRUST (LB) CALCULATION	DEVIATION (%)	
360	9,286	8,288	7,935	- 4.3	
400	10,980	9,800	9,566	- 2.4	
440	13,118	11,708	11,407	- 2.6	
480	15,539	13,869	13,449	- 3.0	
520	18,662	16,656	15,686	- 5.8	
560.	23,639	21,090	18,115	-14.1	
600					
MAXIMUM ENDURANCE	FUEL FLOW	6,200 (LBS/H)			
	THRUST (min)	5,534 (LB)			
	AIR SPEED	240 (KTAS)			
VARIATION OF K	K_1	0.014+0.0001x60 = 0.02			
	K_2	2.0×10^8			

Second, in all drag index ranges, the thrust required is much less than published data at high air speed range (approximately above Mach number 0.8).

To compensate for the first problem, that is low thrust required in high drag index range, it is needed to increase the K_1 value, but the thrust required value is

TABLE 6

THRUST REQUIRED DEVIATION AT DI 80

ALTITUDE	0 FT	GROSS WEIGHT	40,000 LBS	D.I	80
MILITARY	THRUST	21,800 LB	THRUST	FUEL FLOW x 0.9165	
	FUEL FLOW	23,785 LB/H			

AIR SPEED (KTS)	FUEL FLOW (LBS/H)	THRUST (LB) HAND BOOK	THRUST (LB) CALCULATION	DEVIATION (%)
360	10,104	9,261	8,674	- 6.3
400	12,094	11,085	10,479	- 5.5
440	14,487	13,278	12,512	- 5.8
480	17,419	15,965	14,763	- 7.5
520	21,477	19,658	17,228	-12.6
560.				
600				

MAXIMUM ENDURANCE	FUEL FLOW	6,650	(LBS/H)
	THRUST (min)	6,003	(LB)
	AIR SPEED	238	(KTAS)

VARIATION OF K	K_1	0.014+0.001x80 = 0.022
	K_2	2.0×10^8

slightly larger than the published data. The way to compensate in both the low and high drag index range is to increase the slope of the K_1 equation and to decrease the starting value.

The modified equation of K_1 is

$$K_1 = (0.0135 + 0.000132 \times \text{Drag Index})$$

And the change of K_1 equation can be illustrated in Figure 2.11.

The second problem, that the thrust required is much less than the FLIGHT MANUAL data in the speed range higher than Mach number 0.8, can be illustrated as the airplane reaches very high flight speeds, the drag rises in a very rapid fashion due to compressibility. Since the generalized equation for parasite drag does not account for compressibility effects, the actual dr g rise is typified by the dashed line in Figure 2.12.

This phenomenon can be investigated from the viewpoint of drag coefficient. The lift and drag coefficients vary with the Mach number, however, the drag coefficient for a fixed value of lift coefficient does not change until the drag-divergence Mach number is reached, at which time the drag coefficient increases sharply. If wind-tunnel tests of a model of the airplane to be analyzed have been conducted, curves of drag coefficient against Mach number for various values of lift coefficient will be available as shown in Figure 2.13.

Drag data above the divergence Mach number cannot be calculated with precision, and to achieve any accuracy at all, wind-tunnel tests or other empirical data must be relied on. It is noted that even wind-tunnel data may not be reliable in the transonic range, and therefore performance guarantees in the transonic range, based purely on wind-tunnel data and/or calculations, cannot be considered completely dependable.

In most instances, the lift-drag relation may be expressed in an analytic form such as

$$C_D = C_{D_o} + C_L^2 / \pi e AR$$

in which case performance characteristics may be analytically determined if similiar analytical expression for power available can be written.

The equation

$$C_D = C_{D_o} + C_L^2 / \pi e AR$$

applies to most aircraft through the Mach number range of 0 to about 4 and a C_L range from 0 to 0.6, for both trimmed and untrummed conditions, however, C_{D_o} and e

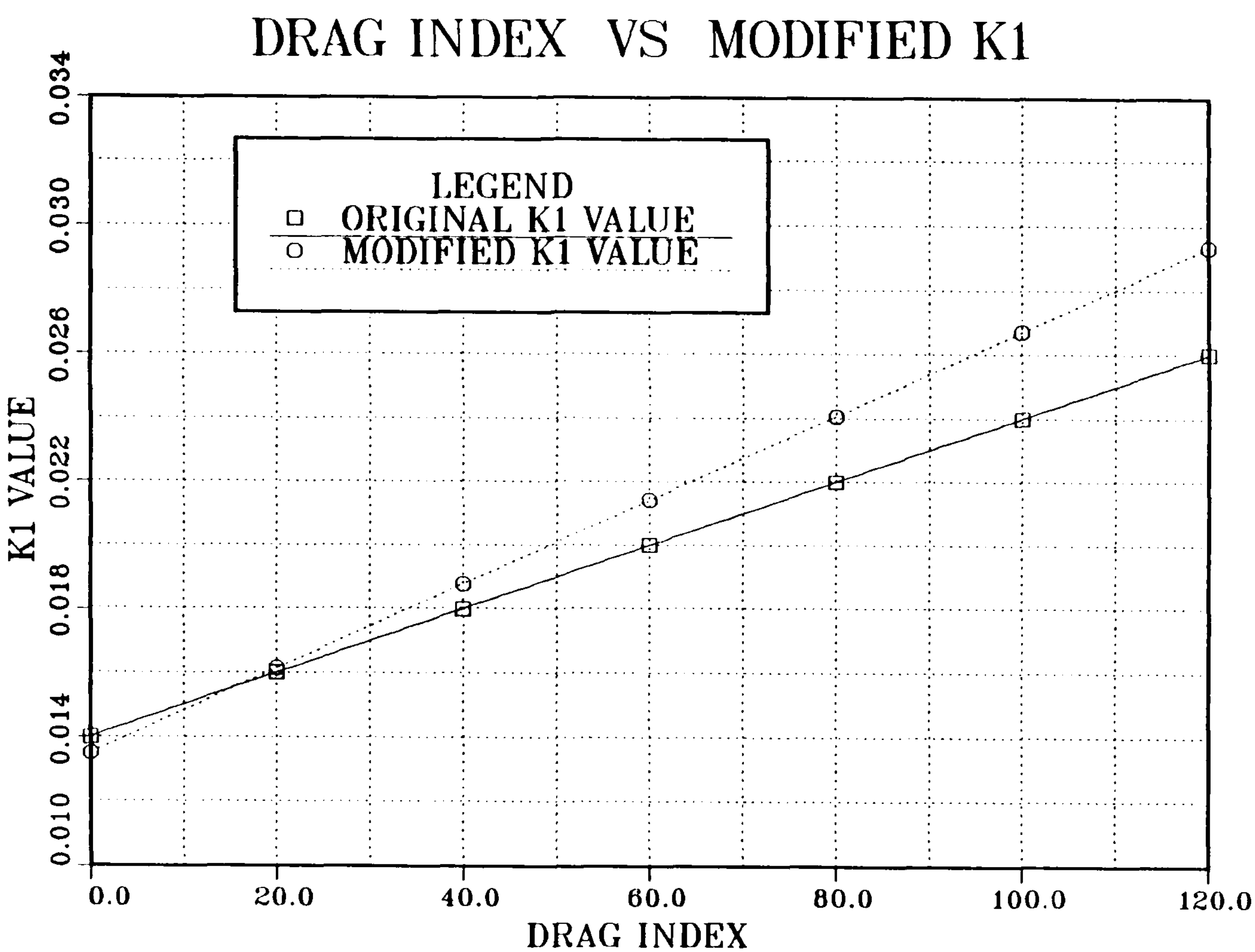

Figure 2.11 Drag Index versus Modified K1.

AIRPLANE DRAG VS VELOCITY

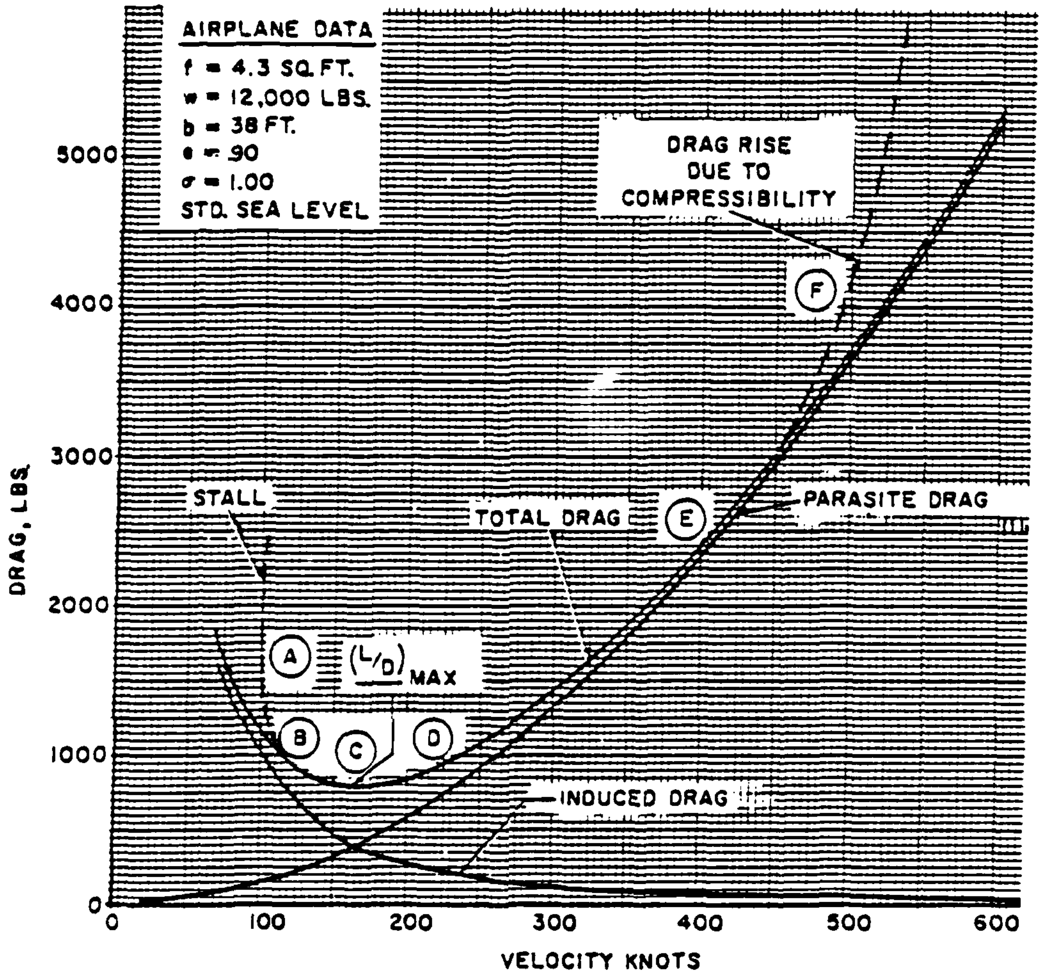

Figure 2.12 Typical Airplane Drag Curve.

become functions of Mach number above Mach numbers of about $M = 0.8$. For supersonic aircraft the value of C_{D_0} increases by a factor of 2 to 3 through the transonic region and thereafter generally exhibits a slight decrease. [Ref. 3:pp. 275-278] The span efficiency factor e shows a steady decrease from $M = 0.8$ to about $M = 2.0$ and thereafter remains almost constant at about 50 to 60% of its low speed value. From $M = 0.8$ to $M = 1.2$ the parasite drag curve quite closely resembles a $\sin^3$ curve, and for approximation analysis may be represented by the following equation

DRAG COEFFICIENT IN TRANSONIC

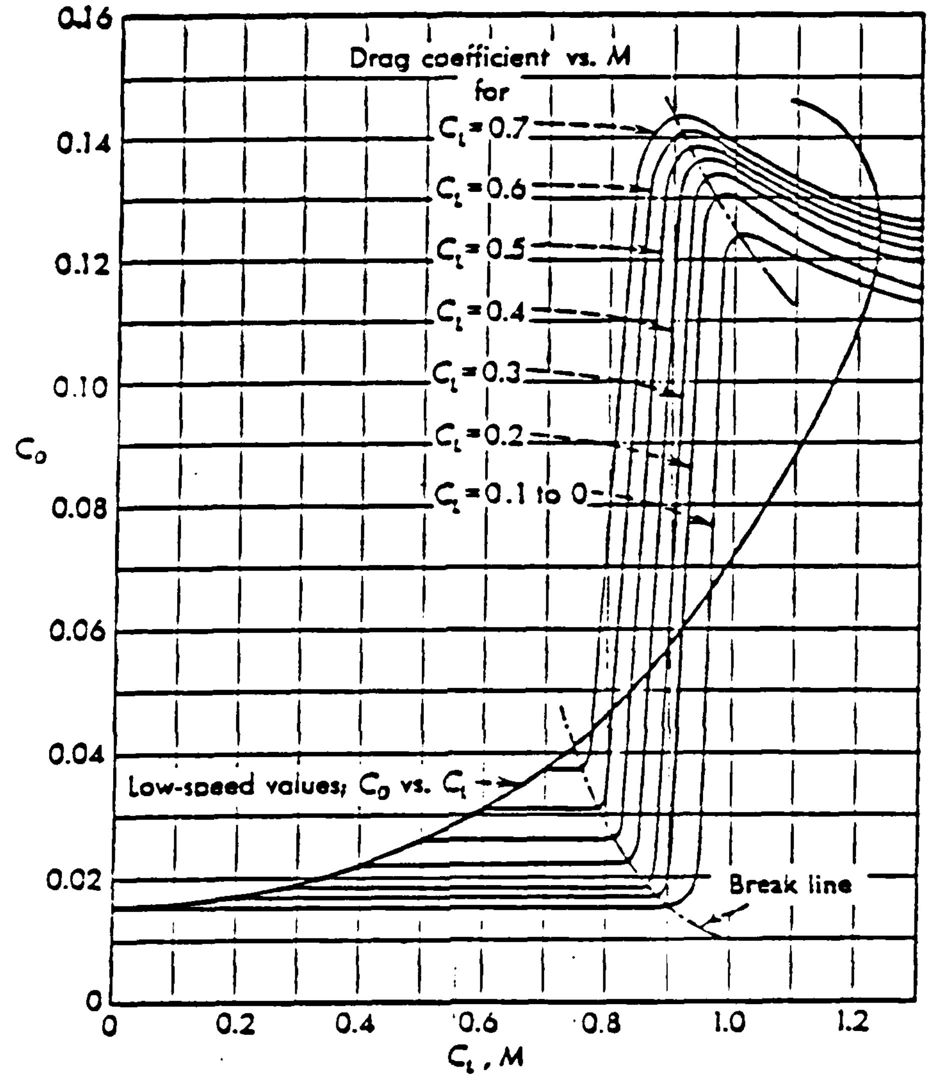

Figure 2.13 Variation of Cd in Transonic region.

$$C_{D_o} = C_{D_{oo}} + (C_{D_{om}} - C_{D_{oo}}) \sin^3 [(M-0.8)/0.8]$$

where $C_{D_{om}}$ = maximum value of parasite drag coefficient

$C_{D_{oo}}$ = minimum low speed value of parasite drag coefficient.

The variation of e may normally be fitted by the polynomial

$$e = e_o - e_2 (M - 0.8)^2 + e_3 (M - 0.8)^3$$

where e_o = low speed value of e

e_2, e_3 = constants which depend on the airplane.

With the above theory and parasite drag equation, it is possible to make the drag equation between the Mach number M.8 to M 1.2.

$$D_o = (1/2) \rho S C_{D_o} V^2$$

at above Mach number M.8, where Mach Drag exist,

$$D_o = (1/2)\rho S [C_{D_{oo}} + (C_{D_{om}} - C_{D_{oo}})\sin^3 \{(M\text{-}0.8)/0.8\}] V^2$$

$$K_1 = (1/2) \rho S C_{D_o}$$

$$D_o = [K_1 + K_1 / C_{D_{oo}} (C_{D_{om}} - C_{D_{oo}}) \sin^3 \{(M\text{-}0.8)/0.8\}] V^2$$

$$= K_1 [1 + (C_{D_{om}} - C_{D_{oo}})/C_{D_{oo}} \sin^3 \{(M\text{-}0.8)/0.8\}] V^2$$

Since K_1 and K_2 are function of air density, it must be considered altitude correction factor of the changing altitude. [Ref. 4]

First consideration is the lapse rate and vertical temperature structure for the standard atmosphere.

The adopted value for the lapse rate($-dT/dH = a$) in the troposphere is the following constant

a) In terms of standard geopotential meters

$$a = 0.0065°C\ m^{-1}$$

b) In terms of standard geopotential feet (a)

$$a = 0.00356616°F\ ft^{-1}$$

c) In terms of the c.g.s. unit of H

$$a = 6.628155 \times 10^{-8}\ °C\ cm^{-2}\ sec^{-2}$$

Accordingly in the troposphere

$$T = T_o - aH \qquad \text{(b)}$$

where a,H and T_o are in consistent unit

Second consideration is the relationship between pressure and vertical displacement.

The following relationships are to be understood as expressed in any system of consistent units.

The air is assumed to obey the perfect gas law, which, in the c.g.s system of units, may be written as

$$\rho = (1/R)(P/T) \quad \text{(c)}$$

where

ρ density of air, gm cm^{-3}

p pressure, dynes cm^{-2}

T absolute temperature,°K

R gas constant for 1 gram of dry air, ergs gm^{-1} $\{°K\}^{-1}$

And the air is assumed also to be in gydrostatic equilibrium and to satisfy the differential equation

$$dp = - g\, dz \quad \text{(d)}$$

where, in c.g.s units,

P pressure, dynes cm^{-2}

ρ density (specific mass), gm cm^{-3}

Z vertical distance, cm

g gravitational acceleration, cm sec^{-2}

The vertical displacement is herein expressed in units of geopotential. Geopotential is defined in differential form by the equation

$$G\, dH = g\, dz \quad \text{(e)}$$

where

Z altitude measured positively upwards at a point

g positive (absolute numerical) value of the acceleration due to gravity at the point

H geopotential at the point

G dimensional constant, the amount of which determines the magnitude of the unit of H in terms of length and time.(the dimension of G are in units of gZ per unit of H.)

Substituting equation (b) in equation (c), equation (e) in equation (d), and combining the results gives

$$dP/P = G/R\,(-dH/T_0 - aH)$$

$$= G/aR\ d(T_0 - aH)/(T_0 - aH)$$

Intergrating the right-hand member between limits of 0 and H gives

$$\ln(P/P_0) = G/aR \ln(T_0 - aH/T_0)$$

Let $\quad n = G/aR$

Then

$$P/P_0 = (T_0 - aH/T_0)^n = (T/T_0)^n \qquad (f)$$

From equation (f), using the consistent numerical values of G,a,and R

$$n = 5.2561 \text{ (dimensionless)}$$

The Figure 2.14 shows the relationship between temperature, pressure, density and altitude.

With several equation investigated in above, the useful relationship between temperature, pressure change and vertical displacement.

For this project, the useful relationships are written as

$$T/T_0 = 1 - 6.875 \times 10^{-6} H$$
$$P/P_0 = (1 - 6.875 \times 10^{-6} H)^{5.2561}$$
$$T = 518.688 \times (1 - 6.875 \times 10^{-6} H)$$
$$a^2 = 1.4 \times 1714.87 \times T$$

Finally the relationship between density of air and vertical displacement is

$$\rho/\rho_0 = (P/RT)/(P_0/R_0T_0)$$
$$= (P/P_0)(T_0/T)$$

And it can be seen by computing several cases that the effect of changing altitude becomes larger than normal density changing ratio.

Let $\quad K_9 = (\rho/\rho_0)^{1.2}$

Then, K_1 is proportional to the K_9 and K_2 is inverse proportional to the K_9 .

As was shown in previous equations, K_2 is proportional to the squared gross weight, and the effect of changing gross weight becomes larger than normal gross weight changing ratio, same as altitude change. Thus, if there was a specific reference value of K_2 for specific gross weight, the relationship between arbitrary gross weight and reference gross weight can be made easily as

SPECIFIC VARIATION VS ALTITUDE

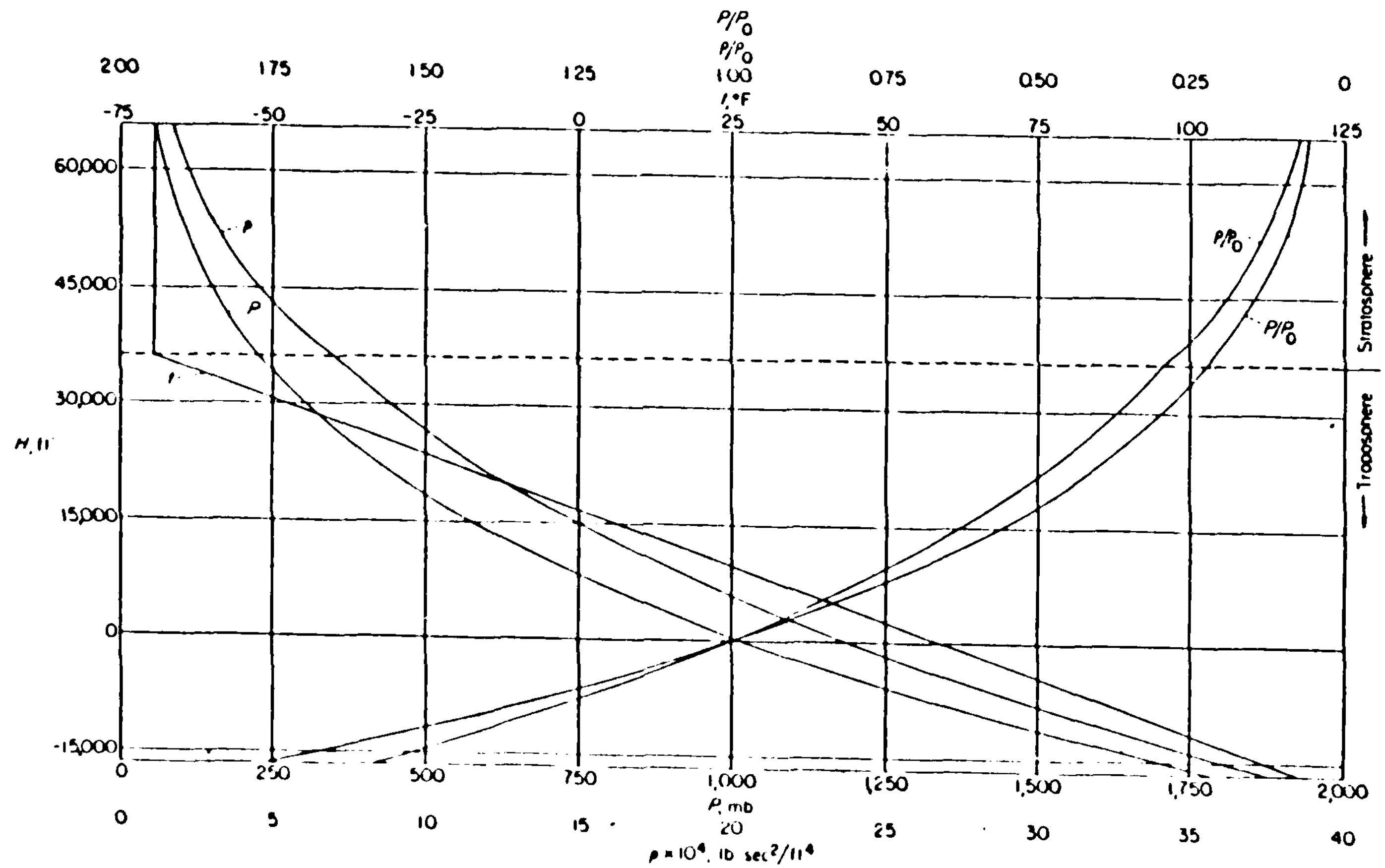

Figure 2.14 Temperature, Pressure, Density against Geopotential H.

$$K_2 = K_{2_{ref}}(W/W_{ref})^{2.7}$$

With given three specific conditions,i.e. altitude, gross weight and drag index, the thrust required can be computed for any airspeed condition with equations shown above. K_1 and K_2 values can be computed with specific altitude, gross weight and drag index, and the thrust required for any air speed can be computed with K_1 and K_2.

The final goal of this project is to compute the fuel flow rate for some given condition, i.e. specific altitude, gross weight, drag index and air speed. The fuel flow is thrust required devided by factor C. The factor C is the value that military power devided by fuel flow of that time, as was shown in previous step.

C = MILITARY THRUST / FUEL FLOW (at military thrust)

5. **Factor C**

To compute factor C, it is necessary to find the relationship military thrust change for different altitude and fuel flow change. This is done in two steps.

a. Computing the military thrust

Altitude is one factor which strongly affects the performance of the turbojet engine. An increase in altitude produces a decrease in density and pressure and, if below the tropopause, a decrease in temperature. If a typical nonafterburning turbojet engine is operated at a constant RPM and true airspeed, the variation of thrust and specific fuel consumption with altitude can be approximated from Figure 2.14. The variation of density in the standard atmosphere is shown by the values of density ratio at various altitudes. Typical values of the density ratio at specific altitudes are as follows

Altitude, ft:	Density ratio
Sea level	1.0000
5,000	0.8617
10,000	0.7385
22,000	0.4976
35,000	0.3099
40,000	0.2462
50,000	0.1532

If the fixed geometry engine is operated at a constant V (TAS) in subsonic flight and constant N (RPM) the inlet velocity, inlet ram, and compressor pressure

ratio are essentially constant with altitude. An increase in altitude then causes the engine air mass flow to decrease in a manner very nearly identical to the altitude density ratio. Of course, this decrease in mass flow will produce a significant effect on the output thrust of the engine. Actually, the variation of thrust with altitude is not quite as severe as the density variation because favorable decreases in temperature occur. The decrease in inlet air temperature will provide a relatively greater combustion gas energy and allow a greater jet velocity. The increase in jet velocity somewhat offsets the decrease in mass flow. Of course, an increase in altitude provides lower temperatures below the tropopause. Above the tropopause, no further favorable decrease in temperature takes place so a more rapid variation of thurst will take place. The approximate variation of thrust with altitude is represented by Figure 2.15 and can be computed with the equation as follows

$$T_{avail} = T_{ssl}\,\delta / \sigma^{(1/2)}$$

where $\delta = P_{alt} / P_{ssl}$

$\sigma = \rho_{alt} / \rho_{ssl}$

Some typical values at specific altitudes are as follows

Altitude, ft:	Ratio of (Thrust at altitude/thrust at SSL)
Sea level	1.0000
5,000	0.8960
10,000	0.8003
15,000	0.7114
20,000	0.6296
25,000	0.5544
30,000	0.4855

Since the change in density with altitude is quite rapid at low altitude, turbojet takeoff performance will be greatly affected at high altitude. Also note that the thrust at 35,000 ft. is approximately 39 percent of the sea level value.

The thrust added by the afterburner of a turbojet engine is not affected so greatly by altitude as the basic engine thrust. The use of afterburner may provide a thrust increase of 50 percent at low altitude or as much as 100 percent at high altitude. [Ref. 2 pp. 119-121]

When the inlet ram and compressor pressure ratio is fixed, the principal factor affecting the specific fuel consumption is the inlet air temperature. When the inlet air temperature is lowered, a given heat addition can provide relatively greater changes in pressure or volume. As a result, a given thrust output requires less fuel flow and the specific fuel consumption is reduced. While the effect of altitude on specific fuel consumption does not compare with the effect on thrust output, the variation is large enough to strongly influence a typical variation of specific fuel consumption with altitude. Generally, the specific fuel consumption decreases steadily with altitude until the tropopause is reached and the specific fuel consumption at this point is approximately 80 percent of the sea level value.

Above the tropopause the temperature is constant and altitude slightly above the tropopause causes no further decrease in specific fuel consumption. Actually, altitudes much above the tropopause bring about a general deterioration of overall engine efficiency and the specific fuel consumption begins an increase with altitude. The extreme altitudes above the tropopause produce low combustion chamber pressures, low compressor Reynolds Numbers, low fuel flow, etc, which are not conducive to high engine efficiency.

The available military thrust for this project with various altitude is shown on Table 7 and Figure 2.15.

TABLE 7

MILITARY THRUST FOR DIFFERENT ALTITUDES

Altitude (ft)	Military thrust (lbs)
Sea level	21,800
5,000	19,533
10,000	17,446
15,000	15,508
20,000	13,725
25,000	12,086
30,000	10,584

The appropriate equation for the curve is as follows

$$\text{THRUST} = 21{,}800 - 0.4 \times \text{ALTITUDE}$$

b. Computing the fuel flow at military thrust

There are three effective factors required to change the fuel flow. One is altitude, another is drag index, and the other is gross weight. To get an appropriate

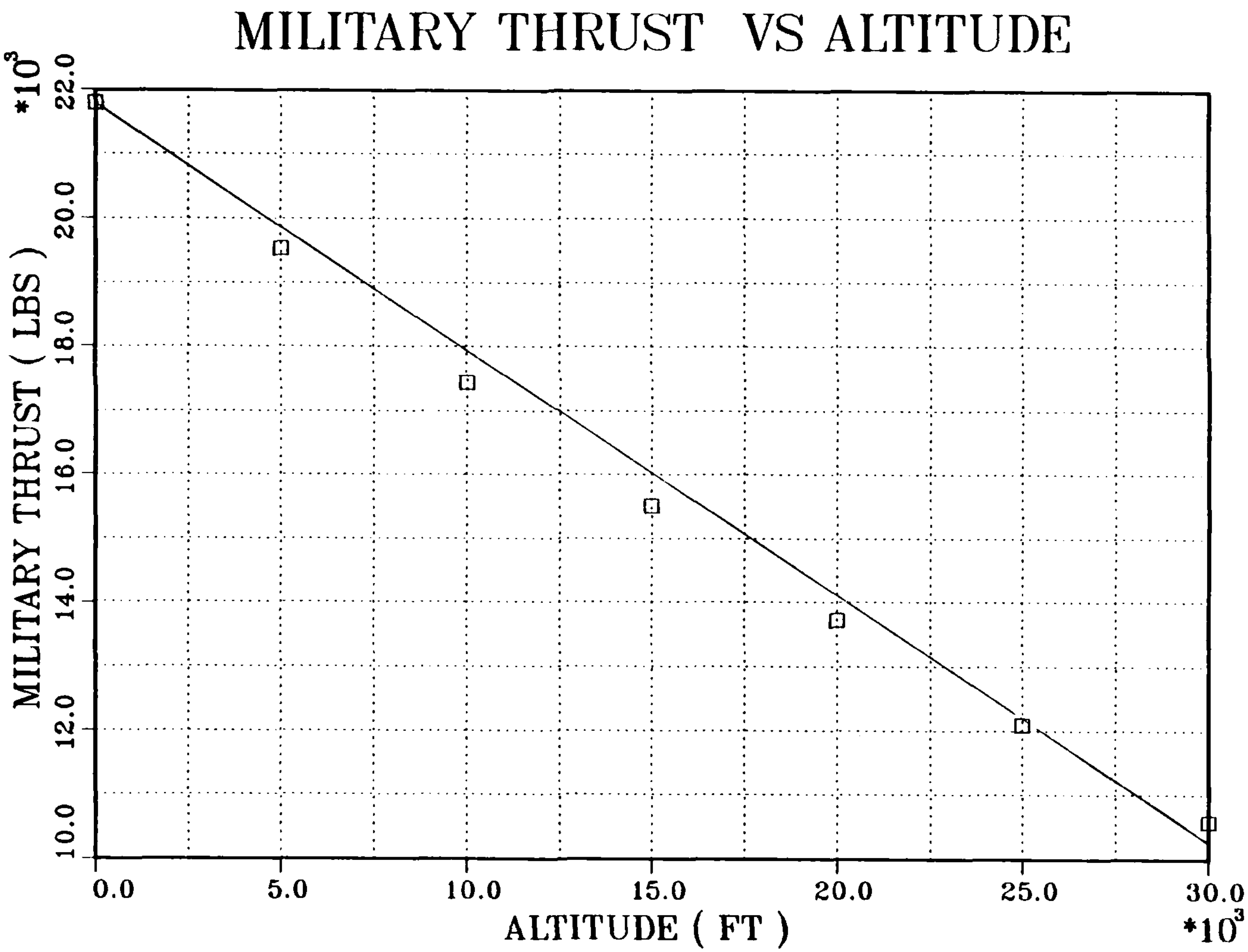

Figure 2.15 Military Thrust against Altitude.

fuel flow equation as a function of altitude, drag index and gross weight, find the relationship between fuel flow and each effective factor, then combine the relationship totally.

(1) *Relationsnip between fuel flow and altitude.*

According to the NATOPS FLIGHT MANUAL data, the fuel flow is decreasing gradually with increasing the altitude with other conditions constant. To find a relationship between fuel flow and altitude, specific drag index and gross weight are fixed. to select the available data. Table 8 and Figure 2.16 show the relationship when drag index and gross weight are fixed to zero and 40,000 lbs.

After curve-fitting, the fuel flow equation stated as follows

F_1 = 26,000 - 0.64 x Altitude (at below 8,000 ft)

F_1 = 20,840 - 0.37 x (Altitude - 8,000) (at above 8,000 ft)

where

F_1 = Fuel flow considered with different altitude

(2) *Relationship between fuel flow and drag index.*

According to the NATOPS FLIGHT MANUAL data, the fuel flow is decreased as the drag index increased. As was considered in step (1), a specific altitude and gross weight will be fixed to find the relationship between fuel flow and drag index.

Table 9 and Figure 2.17 show the relationship when altitude and gross weight are fixed to zero and 40,000 lbs.

To find the relationship between fuel flow and drag index with different altitude, the computed fuel flow from step (1) must be used. With above two variable coefficients, the fuel flow equation is

F_2 = F_1 - 16.65 x Drag Index (at below Drag Index 20)

F_2 = (F_1 + 333.3) - 22.1 x Drag Index (at above Drag Index 20)

where

F_1 = Fuel flow came from step (1)

F_2 = Fuel flow considered with different altitude and drag index

(3) *Relationship between fuel flow and gross weight.*

According to the NATOPS FLIGHT MANUAL, the fuel flow decreased a small amount with increasing gross weight. To find the relationship

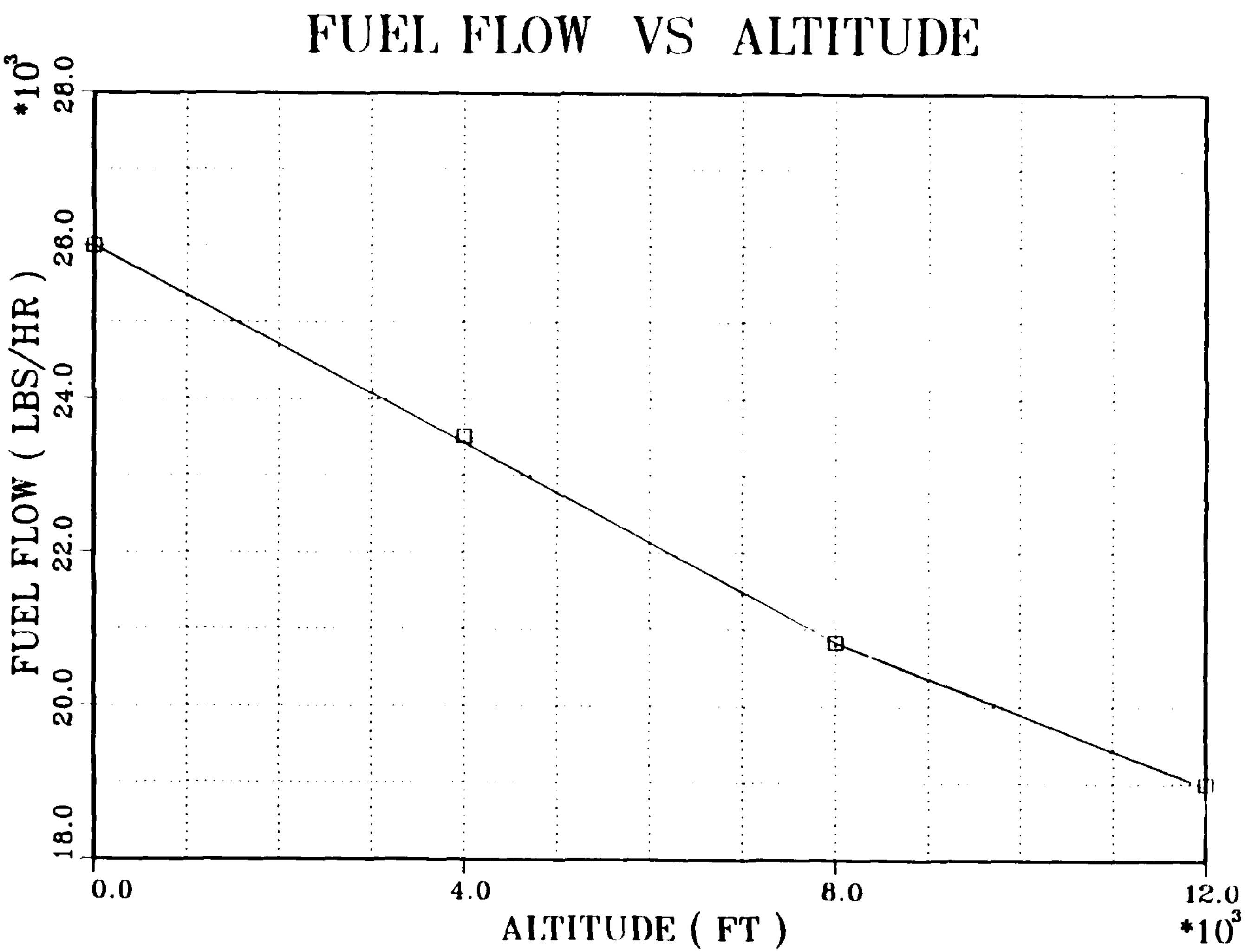

Figure 2.16 Relationship between Fuel Flow and Altitude.

TABLE 8

FUEL FLOW FOR DIFFERENT ALTITUDES

Altitude (ft)	Fuel Flow (lbs/hr)
Sea level	26,000
4,000	23,520
8,000	20,840
12,000	19,000

TABLE 9

FUEL FLOW FOR DIFFERENT DRAG INDEXS

Drag Index	Fuel Flow
0	26,000
20	25,680
40	25,110
60	24,425
80	23,785
100	23,225
120	22,745

TABLE 10

FUEL FLOW FOR DIFFERENT GROSS WEIGHTS

Grossweight (lbs)	Fuel Flow (lbs/hr)
35,000	26,000
40,000	26,000
45,000	25,995
50,000	25,990
55,000	25,985
60,000	25,980

between fuel flow and gross weight, a specific altitude and drag index must be selected as in the previous steps. Table 10 and Figure 2.18 show the relationship when altitude and drag index are fixed to standard sea level and zero.

It can be seen from Figure 2.18 that the relationship between fuel flow and gross weight is linear. The final equation of fuel flow with different three variables, i.e., altitude, drag index and gross weight, can be written as follows

$F_3 = F_2 - 0.002x(\text{Gross weight} - 40{,}000)$ (at heavier than 40,000lbs)

$F_3 = F_2$ (at lighter than 40,000 lbs)

where

$F_2 =$ Fuel Flow came from step (2)

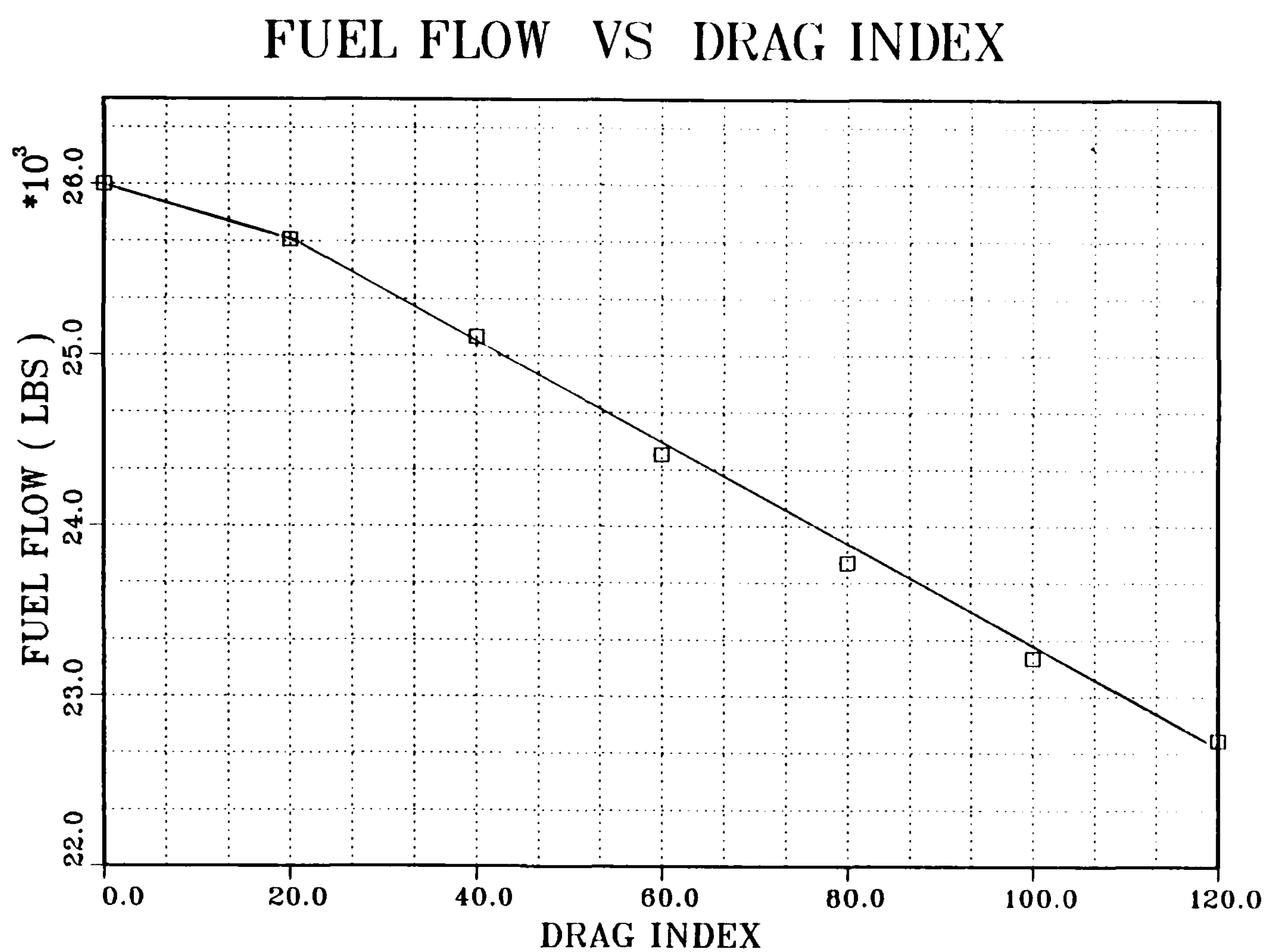

Figure 2.17 Fuel Flow versus Drag Index.

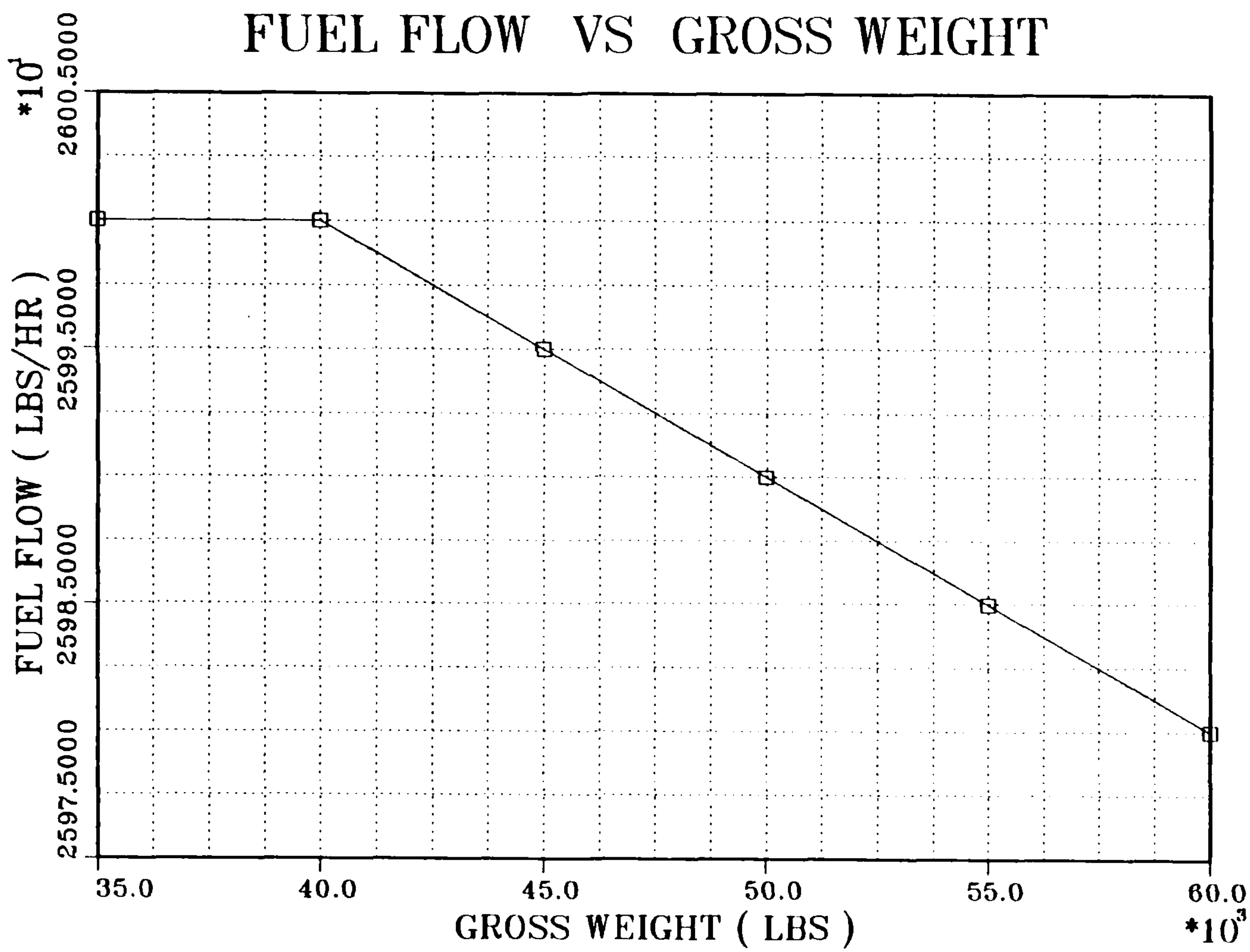

Figure 2.18 Fuel Flow versus Gross Weight.

F_3 = Fuel Flow considered with different altitude, drag index and gross weight

Now the military thrust and fuel flow for any altitude, drag index and gross weight can be computed with the above step (1) and (2). The desired factor C can be computed with the military thrust and fuel flow.

Factor C = Military thrust/ Fuel Flow

To compute the fuel flow for any various specific condition is the final goal of this project. The fuel flow can be computed with thrust required, i.e. total drag, devided by factor C. The total drag and factor C had been computed in previous step already.

FUEL FLOW = THRUST REQUIRED / FACTOR C

With the above equation, it is possible to find the required fuel flow for different specific conditions without interpolating the flight manual data.

III. RESULT

A. BASE LINE CONDITION

Given condition

Gross weight	40,000 lbs
Drag Index	20
Altitude	Standard Sea Level

1. Converting velocity versus fuel flow to velocity versus drag

Military thrust (at sea level) 21,800 lb

Fuel flow (at military thrust) 25,680 lb/hr

Thrust = Fuel Flow x Factor C

C = Military thrust / Fuel Flow

at military thrust

C = 21,800/25,680 = 0.84891

The thrust required for each velocity can be computed by fuel flow times factor C(0.84891). The Table 11 shows the result.

TABLE 11

THRUST REQUIRED FOR EACH SPEED

VELOCITY(KTAS)	FUEL FLOW(LBS/H)	THRUST REQUIRED(LBS)
360	7,725	6,558
400	9,014	7,652
440	10,623	9,018
480	12,462	10,579
520	14,652	12,438
560	17,377	14,752
600	21,393	18,160
MIL	25,680	21,800

2. Computing the minimum thrust and maximum endurance speed.

Maximum endurance speed can be obtained from maximum endurance Mach number graph in NAPTOPS FLIGHT MANUAL (see Figure 2.7). The maximum endurance Mach number is M 0.378. And maximum endurance fuel flow can be

obtained from the maximum endurance fuel flow graph in NATOPS MANUAL (see Figure 2.8).

The minimum thrust can be obtained from the maximum endurance fuel flow by multiplying with the factor C computed in step 1.

maximum endurance fuel flow ; 5,600 lb/hr

$T_{min} = 5{,}600 \times 0.84891 = 4{,}753.9$ lb

3. **Computing the K_1**

At maximum endurance speed, the thrust required is equal to the total drag, that is twice the induced drag or parasite drag, i.e. induced drag and parasite drag are the same at this point.

$$T_{min} = D_{total} = 2D_i = 2D_o$$

$$D_o = C_{D_o}(1/2)\rho V^2 A$$

where $D_o = 4{,}753.9 / 2 = 2{,}376.9$ lb

$\rho = 0.0023769$ lb sec^2/ft^4

$V = 0.378 \times 1{,}116.4$ ft/sec

$A = 530$ ft^2

$$\begin{aligned} C_{D_o} &= 2D_o / (\rho V^2 A) \\ &= (4{,}753.9)/(0.0023769 \times 422.18^2 \times 530) \\ &= 0.02117183 \end{aligned}$$

$$\begin{aligned} D_o &= (0.02117183)(1/2)(0.0023769)(530)V^2 \\ &= 0.01333\ V^2 \end{aligned}$$

Thus $K_1 = 0.0133333$

4. **Computing the K_2**

Use the same concept as for K_1

$$T_{min} = D_{total} = 2D_i = 2D_o$$

$$D_o = 2W^2 / (\rho A V^2)(1/\pi e AR)$$

$$\begin{aligned} 1/\pi e AR &= 2W^2 / (\rho A V^2 D_i) \\ &= (0.0023769)(530)(422.18)^2(2376.9)/(2 \times 40000^2) \\ &= 0.16677964 \end{aligned}$$

$$D_i = (2W^2/\rho A)(1/\pi eAR)(1/V^2)$$
$$= (2 \times 40{,}000^2 / 0.0023769 \times 530)(0.16677964)(1/V^2)$$

Thus

$$K_2 = 4.2365 \times 10^8$$
$$D_{total} = \text{Thrust required} = K_1V^2 + K_2/V^2$$

These K_1 and K_2 values are modified a little to fit the curve of hand book data by the trial and error method introduced in Chapter 2. Thus the thrust required equation can be written as

$$D = 0.01614V^2 + 2.0 \times 10^8/V^2$$

But if there were no consideration about the Mach effects, the drag at high Mach number, i.e. greater than M 0.8, will be much less than the hand book data. C_D becomes a function of Mach number above Mach numbers of about M 0.8. In this region, the parasite drag curve quite closely resembles a $\sin^3$ curve, and for approximation analysis may be represented by the equation

$$C_{D_o} = C_{D_{oo}} + (C_{D_{om}} - C_{D_{oo}}) \sin^3 \{(M - 0.8)/0.8\}$$

Thus the drag equation between the Mach number M 0.8 and M 1.2

$$D_o = 1/2\, \rho S C_{D_o} V^2$$
$$D_o = 1/2\, \rho S\, [C_{D_{oo}} + (C_{D_{om}} - C_{D_{oo}}) \sin^3\{(M - 0.8)/0.8\}]\, V^2$$
$$K_1 = 1/2\, \rho S C_{D_o}$$
$$D_o = [K_1 + K_1/C_{D_{oo}} (C_{D_{om}} - C_{D_{oo}}) \sin^3\{(M - 0.8)/0.8\}]\, V^2$$
$$= K_1[1 + (C_{D_{om}} - C_{D_{oo}})/C_{D_{oo}} \sin^3\{(M - 0.8)/0.8\}]\, V^2$$

where $C_{D_{om}} - C_{D_{oo}} = 0.55$

$C_{D_{oo}} = 0.018$

$$D_o = K_1[1 + 0.55/0.018 \sin^3 \{(M - 0.8)/0.8\}]V^2$$

5. **Converting to fuel flow from drag**

Fuel flow for each velocity can be computed from thrust required, i.e. total drag, devided by factor C.

C = MILITARY THRUST FUEL FLOW (at military thrust)

Military thrust and fuel flow can be computed with the equation that found in Chapter 2.

Military thrust = 21,800 - 0.4 x Altitude
= 21,800 - 0.4 x 0
= 21,800

Fuel flow;

F_1 = 26,000 - 0.64 x Altitude
= 26,000 - 0.64 x 0
= 26,000

F_2 = F_1 - 16.65 x Drag Index
= 26,000 - 16.65 x 20
= 25,667

F_3 = F_2 - 0.002 x (Gross weight - 40,000)
= 26,667 - 0.002 x (40,000 - 40,000)
= 25,667

Thus C = 21,800/25,667
= 0.84934

Fuel flow = Thrust required / 0.84934

Table 12 and Figure 3.1 show the results of the calculation and deviation from hand book data.

B. VARIATION NUMBER 1 - GROSS WEIGHT

Given condition

Gross weight	50,000 lbs
Drag Index	20
Altitude	Standard Sea Level

TABLE 12

THRUST REQUIRED AND FUEL FLOW AT BASE LINE CONDITION

ALTITUDE	O FT		GROSS WEIGHT	40,000 LBS	DRAG INDEX	20
MILITARY	THRUST	21,800 LB	THRUST		FUEL FLOW x 0.84891	
	FUEL FLOW	25,680 LBS/H				
AIR SPEED	FUEL FLOW (LBS/H)		THRUST REQUIRED (LB)		DEVIATION (%)	
(KTAS)	HAND BOOK	CALCULATION	HAND BOOK	CALCULATION	THRUST	FUEL FLOW
360	7,725	7,662	6,557	6,508	-0.4	-0.8
400	9,014	9,188	7,652	7,804	2.0	1.9
440	10,623	10,920	9,018	9,275	2.8	2.8
480	12,462	12,847	10,579	10,911	3.1	3.1
520	14,652	14,962	12,438	12,708	2.2	2.1
560	17,377	17,371	14,751	14,754	-0.0	-0.0
600	21,393	21,201	18,160	18,007	-0.8	-0.9
MAXIMUM ENDURANCE		FUEL FLOW	5,600			(LBS/H)
		THRUST (MINIMUM)	5,600x0.84891 = 4,754			(LB)
		AIR SPEED	250			(KTAS)

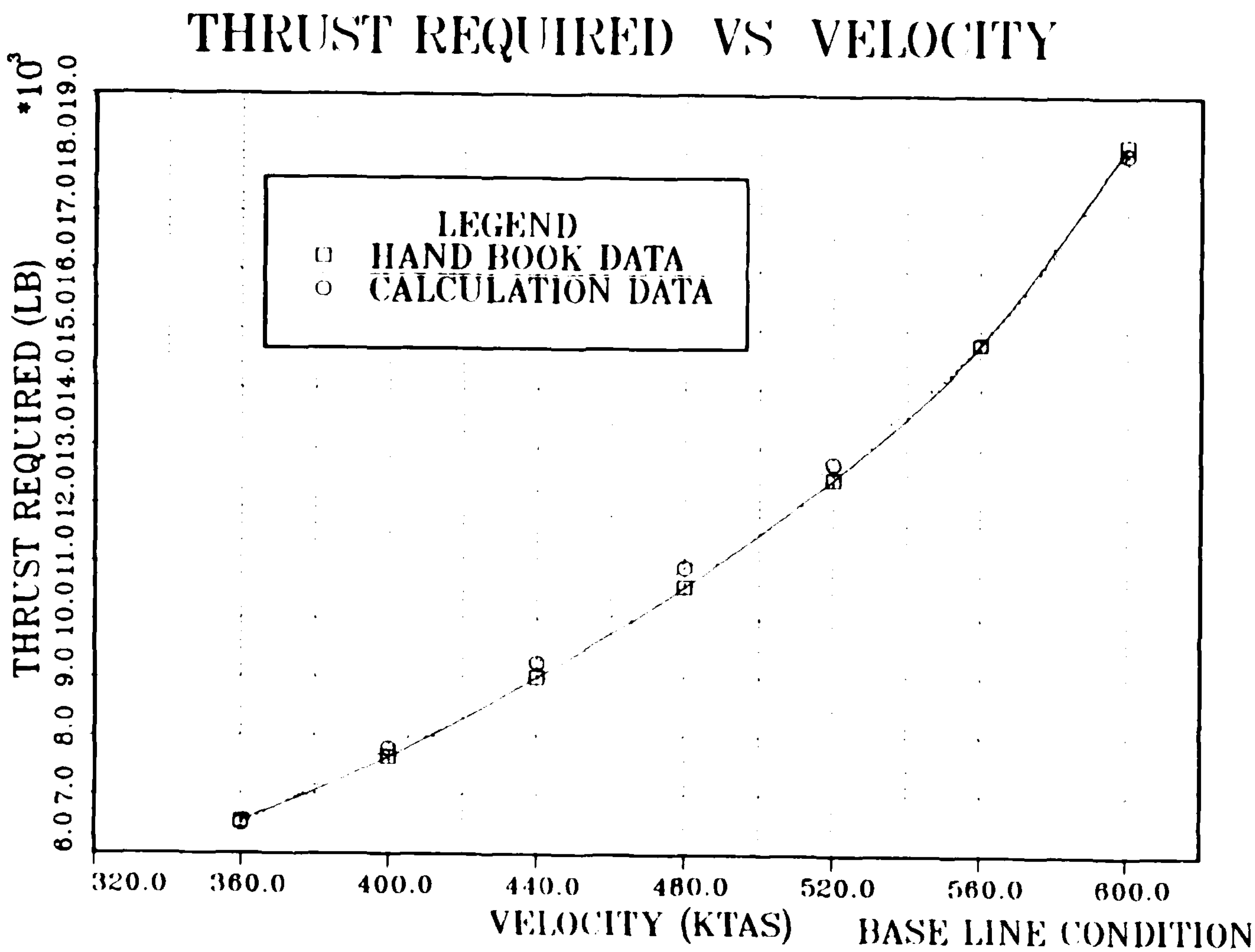

Figure 3.1a Thrust Required at Base Line condition.

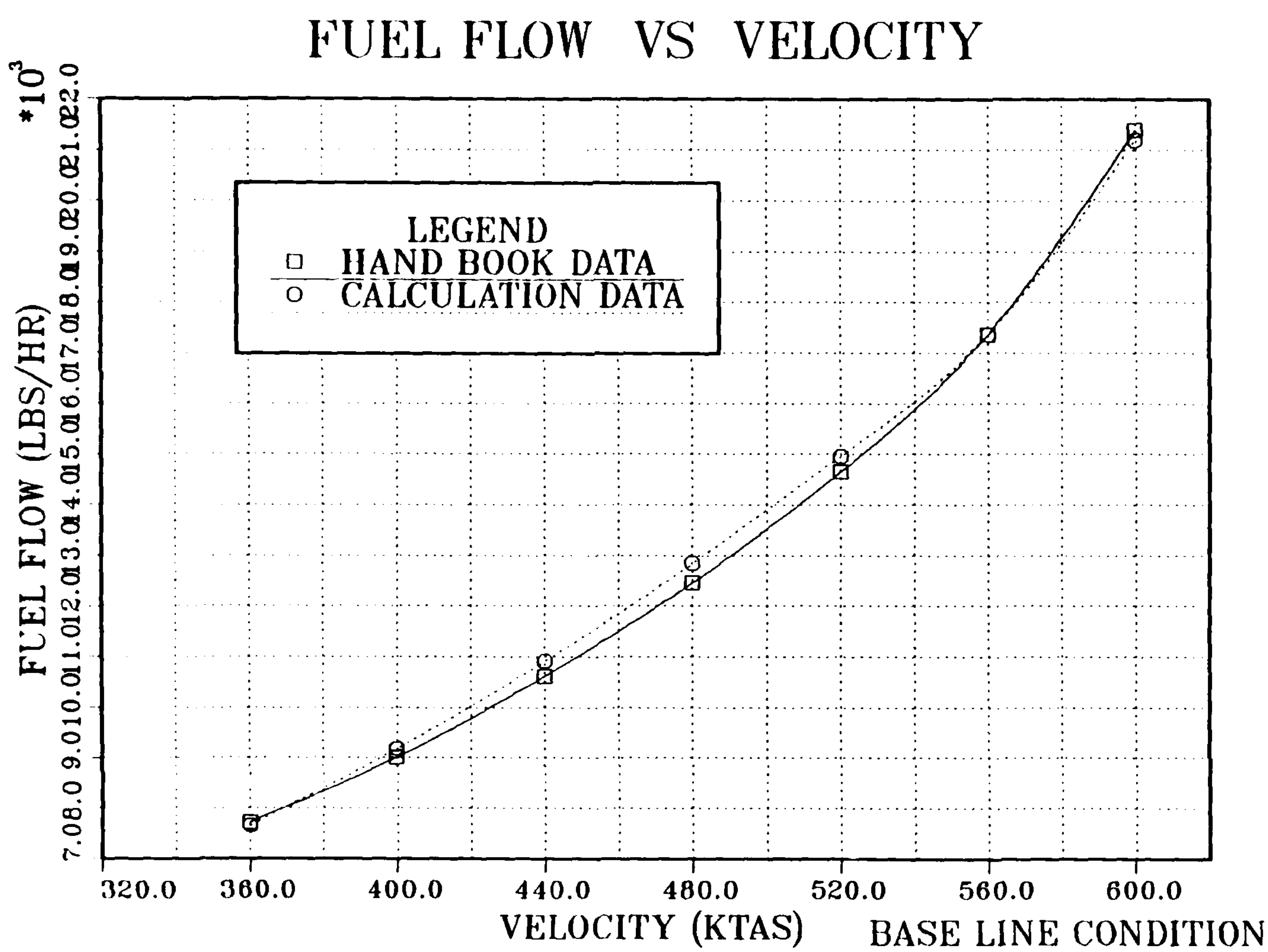

Figure 3.1b Fuel Flow at Base Line condition.

The gross weight is changed only from the oase line condition. It is necessary to consider the change of K_1 and K_2.

$$K_1 = 1/2\, \rho C_{D_o} S$$

As you can see from the above equation, K_1 is independent of gross weight. Thus there was no change in K_1.

But K_2 is a function of gross weight (as discussed in chapter 3).

$$K_2 = 2W^2/(\rho\ S\ \pi\ e\ AR)$$

It can be seen that the K_2 is proportional to the squared gross weight, but the effects of varing gross weight become more largely.

Thus $K_{2_{50,000lb}} = K_{2_{40,000lb}} (50,000/40,000)^{2.7}$

$$= 2.0\text{x}10^8 \text{ x } (1.25)^{2.7}$$

$$= 3.653 \text{ x } 10^8$$

$$D = 0.0164V^2 + 3.653\text{x}10^8/V^2$$

Military thrust = 21,800 lb

Fuel flow;

F_1 = 26,000 - 0.64 x Altitude
= 26,000

F_2 = F_1 - 16.65 x Drag index
= 26,000 - 16.65 x 20
= 25,667

F_3 = F_2 - 0.002 x (Gross weight - 40,000)
= 25,667 - 0.002 x (50,000 - 40,000)
= 25,647

C = 21,800/25,647
= 0.85

Fuel flow = Thrust required / 0.85

Table 13 and Figure 3.2 show the results of calculation and deviation from hand book data.

TABLE 13

THRUST REQUIRED AND FUEL FLOW WITH GROSS WEIGHT VARISTION

ALTITUDE	0 FT	GROSS WEIGHT	50,000 LBS		DRAG INDEX	20
MILITARY	THRUST	21,800 LB	THRUST	FUEL FLOW x 0.85023		
	FUEL FLOW	2[illegible],640 LBS/H				
AIR SPEED	FUEL FLOW (LBS/H)		THRUST REQUIRED (LB)		DEVIATION (%)	
(KTAS)	HAND BOOK	CALCULATION	HAND BOOK	CALCULATION	THRUST	FUEL FLOW
360	8,137	8,182	6,918	6,954	0.5	0.5
400	9,349	9,607	7,949	8,116	2.7	2.7
440	10,863	11,264	9,236	9,574	3.6	3.7
480	12,718	13,133	10,813	11,163	3.2	3.3
520	14,840	15,203	12,617	12,922	2.4	2.4
560	17,582	17,575	14,949	14,938	-0.1	-0.0
600	21,656	21,374	18,413	18,168	-1.3	-1.3
MAXIMUM ENDURANCE		FUEL FLOW	6,600			(LBS/H)
		THRUST (MINIMUM)	6,600x0.85023 = 5,611			(LB)
		AIR SPEED	281			(KTAS)

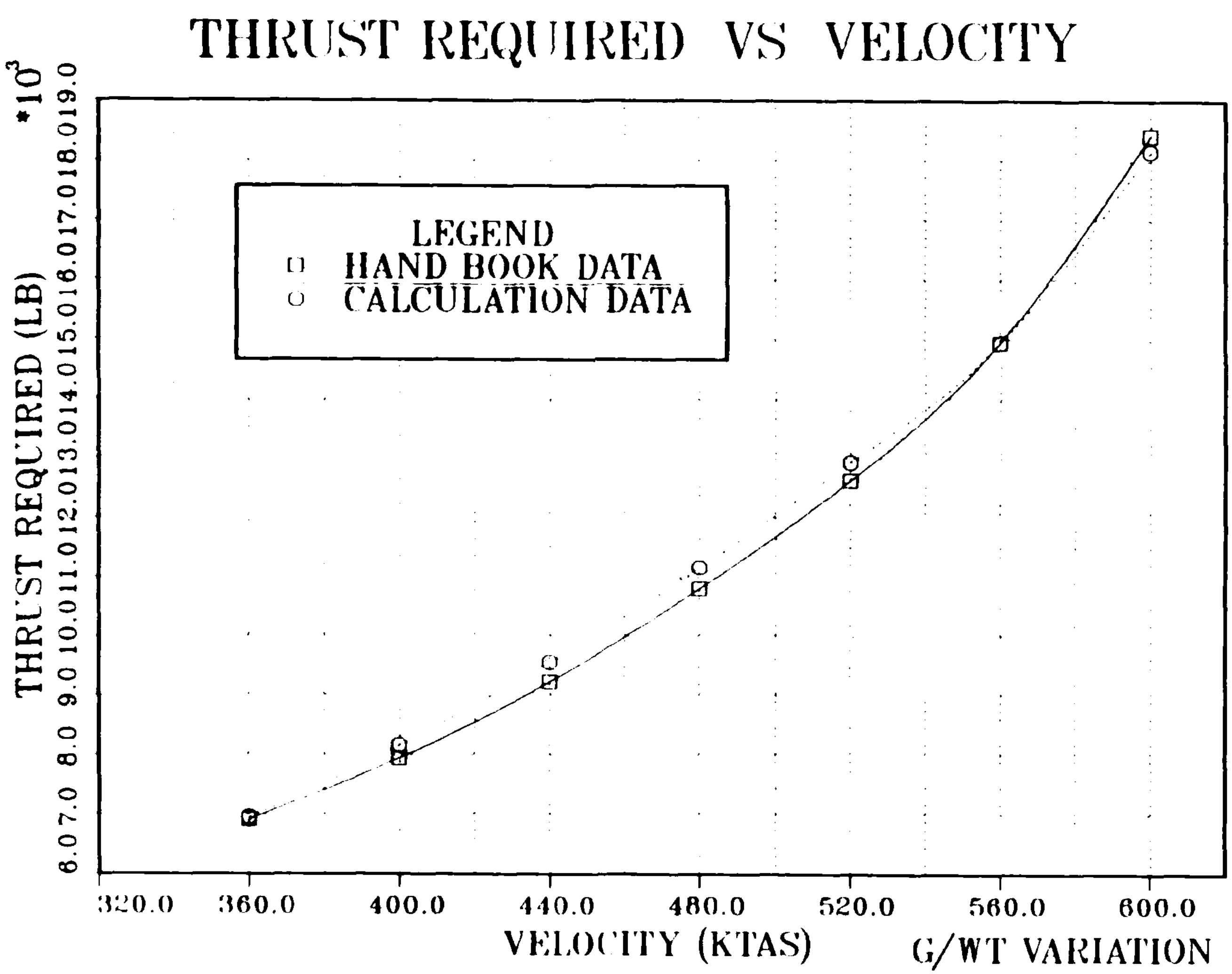

Figure 3.2a Thrust Required with Gross Weight variation.

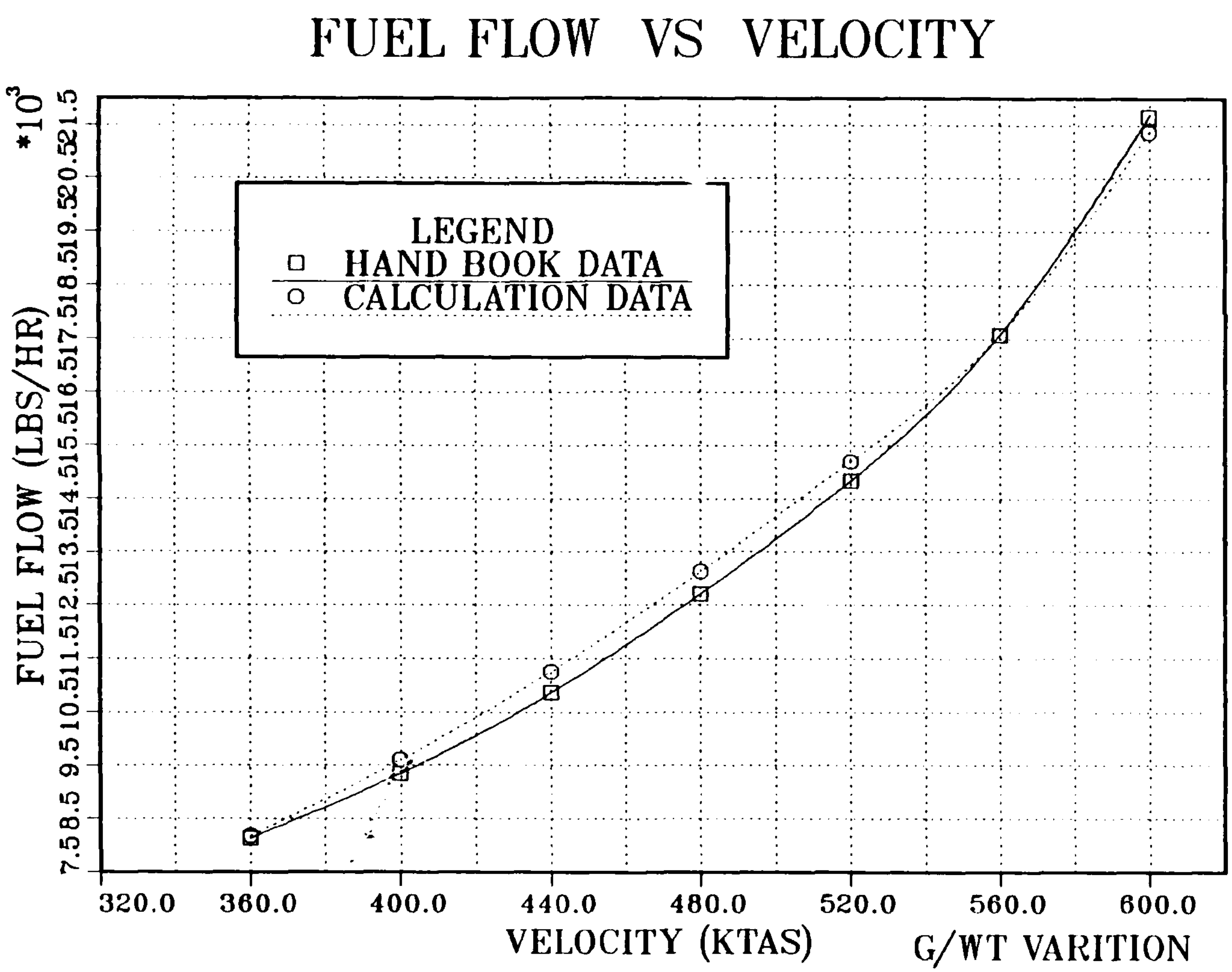

Figure 3.2b Fuel Flow with Gross Weight variation.

C. VARIATION NUMBER 2 - DRAG INDEX

Given condition

Gross weight 40,000 lbs

Drag Index 40

Altitude Standard Sea Level

If the drag index was changed from base line condition, what will be effect on the result? It is necessary to consider the K_1 and K_2 equation.

$$K_1 = 1/2\ \rho S C_{D_o}$$

$$K_2 = 2W^2 / (\rho S \pi e AR)$$

As can be seen, K_2 is a function of gross weight, altitude, wing area and aspect ratio, not of drag index, thus K_2 was unchanged. But K_1 is a function of altitude, wing area and drag index. Thus the appropriate K_1 value can be computed with equation

$$K_{1_{DI40}} = K_{1_{DI20}} \times (K_{1_{DI40}}/K_{1_{DI20}})$$

Now it is necessary to find the relationship of K_1 between drag index 20 and drag index 40. Furthermore, if the relationship of K_1 between any drag index and reference drag index, the various K_1 for different drag index can be computed. To figure out the relationship of K_1 for each drag index, let us compute each K_1 value. The computing method is the same as introduced in the base line condition step.

$$T_{min} = 2D_o$$

$$D_o = 1/2\ \rho S C_{D_o} V^2$$

$$C_{D_o} = 2D_o/(\rho S V^2)$$

$$K_1 = 1/2\ \rho S C_{D_o}$$

For example

when DI = 0

$$D_o = T_{min}/2 = 4{,}443.8/2 = 2{,}221.9$$

$$C_{D_o} = 2(2{,}221.9)/(0.0023769 \times 530 \times 438.9^2) = 0.018132$$

Thus

$$K_1 = 1/2\ \rho S C_{D_o}$$

$= 1/2\ (0.0023769 \times 530 \times 0.018132)$
$= 0.011534$

Table 14 and Figure 3.3 show the result and the relationship of K_1 for different drag index.

TABLE 14
K1 VALUE VERSUS DRAG INDEX

DI	C	MAXIMUM ENDURANCE FUEL FLOW	MAXIMUM ENDURANCE KTAS	THRUST (LB) MINIMUM	C_{D_O}	K_1
0	0.8385	5,300	260	4,444	0.018	0.012
20	0.8489	5,600	250	4,754	0.021	0.013
40	0.8682	5,920	247	5,140	0.024	0.015
60	0.8925	6,250	241	5,578	0.027	0.017
80	0.9165	6,520	238	5,976	0.029	0.018
100	0.9386	6,800	233	6,383	0.033	0.021
120	0.9584	7,050	230	6,757	0.036	0.022

The K_1 of the base line condition was modified to fit the drag index. As you can see in Figure 3.3 and the investigation in Chapter 2, the relationship between K_1 and drag index is linear and the equation of K_1, function of drag index is

$K_1 = 0.0135 + 0.000132 \times \text{Drag Index}$

Thus

$K_{1_{DI40}} = 0.0135 + 0.000132 \times 40$
$= 0.01878$

Thrust required $= 0.01878V^2 + 2\text{x}10^8/V^2$

Military thrust = 21,800 lb

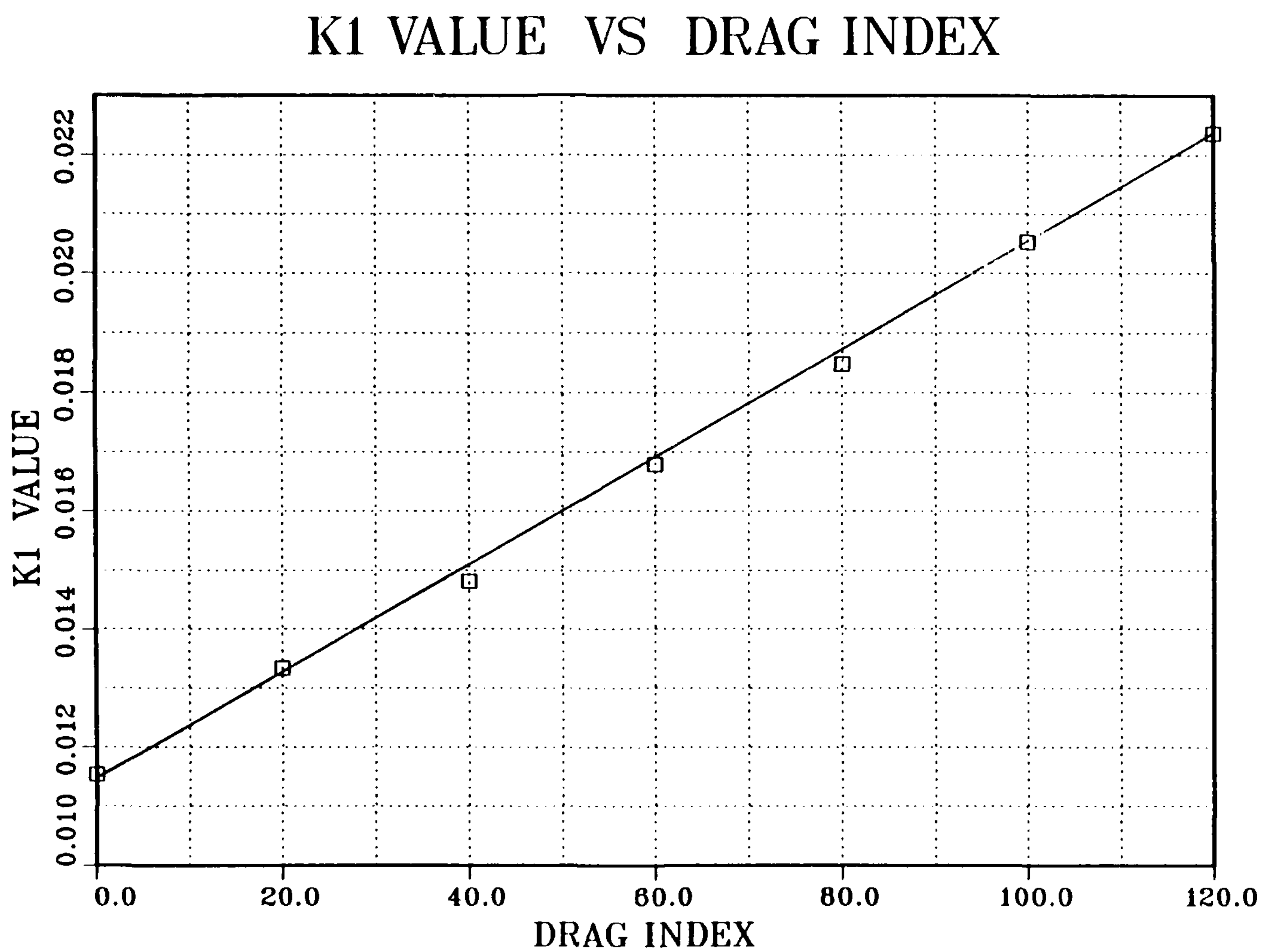

Figure 3.3 K1 value versus Drag Index.

Fuel flow;

$$F_1 = 26{,}000$$

$$\begin{aligned} F_2 &= (F_1 + 333.3) - 22.1 \times \text{Drag Index} \\ &= (26{,}000 + 333.3) - 22.1 \times 40 \\ &= 25{,}449.3 \end{aligned}$$

$$\begin{aligned} F_3 &= F_2 - 0.002 \times (40{,}000 - 40{,}000) \\ &= 25{,}449.3 \end{aligned}$$

$$\begin{aligned} C &= 21{,}800/25{,}449.3 \\ &= 0.8566 \end{aligned}$$

Fuel flow = Thrust required/0.8566

Table 15 and Figure 3.4 show the results of calculation and deviation from hand book data.

D. VARIATION NUMBER 3 - ALTITUDE

Given condition

Gross weight	40,000 lbs
Drag Index	20
Altitude	4,000 ft

If the altitude will be changed only from the base line condition, how will it affect the K_1 and K_2?

The equations of K_1 and K_2 are

$$K_1 = 1/2\, \rho S C_{D_o}$$

$$K_2 = 2W^2/(\rho S \pi e AR)$$

As can be seen in the above equation, the K_1 and K_2 both are functions of the air density, and air density is a function of altitude. Thus the K_1 and K_2 must be changed with varying altitude.

The useful relationships came from the ICAO Report written as

$$\rho = P/RT$$
$$T/T_o = 1 - 6.875\text{x}10^{-6}H$$
$$P/P_o = (1 - 6.875\text{x}10^{-6}H)^{5.2561}$$

TABLE 15

THRUST REQUIRED AND FUEL FLOW WITH DRAG INDEX VARIATION

ALTITUDE	O FT	GROSS WEIGHT	40,000 LBS		DRAG INDEX	40
MILITARY	THRUST	21,800 LB	THRUST		FUEL FLOW x 0.86818	
	FUEL FLOW	25,110 LBS/H				
AIR SPEED	FUEL FLOW (LBS/H)		THRUST REQUIRED (LB)		DEVIATION (%)	
(KTAS)	HAND BOOK	CALCULATION	HAND BOOK	CALCULATION	THRUST	FUEL FLOW
360	8,492	8,736	7,373	7,483	1.5	3.2
400	10,000	10,517	8,682	9,009	3.8	5.2
440	11,879	12,530	10,313	10,733	4.1	5.5
480	13,910	14,763	12,076	12,646	-0.5	6.1
520	16,480	17,212	14,308	14,744	3.0	4.4
560	19,688	19,998	17,093	17,130	0.2	1.6
600						
MAXIMUM ENDURANCE		FUEL FLOW	5,920			(LBS/H)
		THRUST (MINIMUM)	5,920 x 0.86818 = 5,140			(LB)
		AIR SPEED	247			(KTAS)

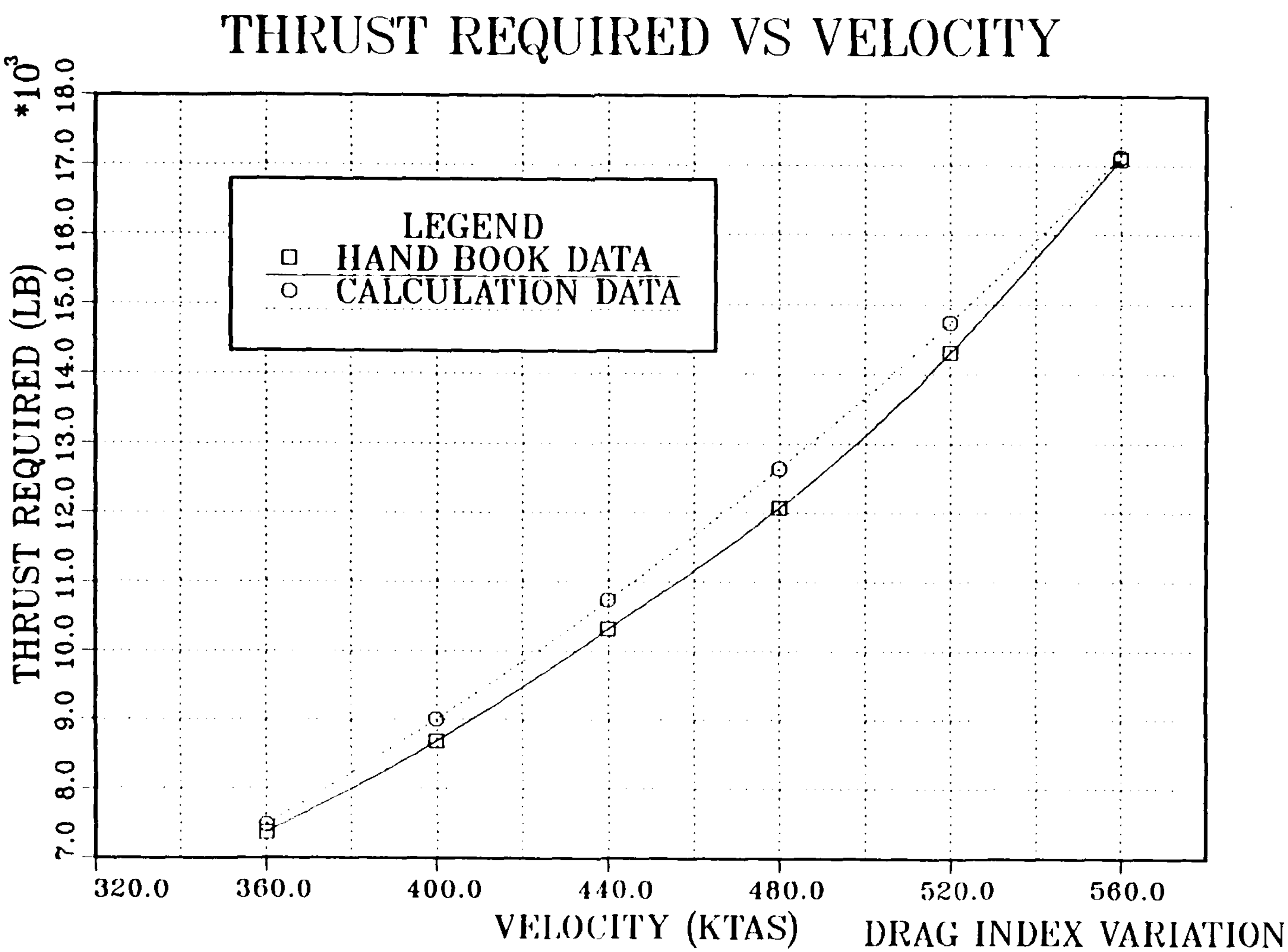

Figure 3.4a Thrust Required with Drag Index variation.

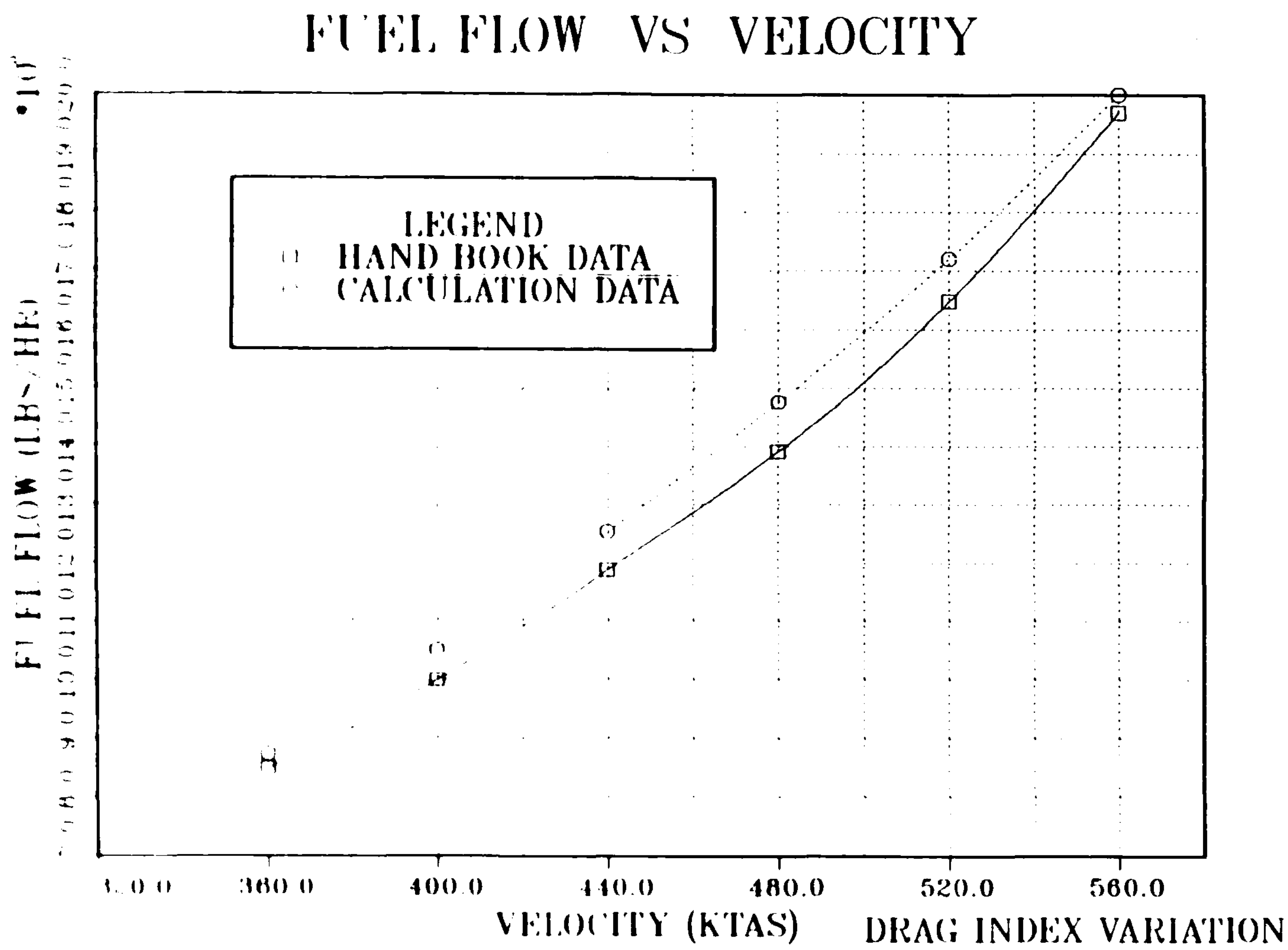

Figure 3.4b Fuel Flow with Drag Index variation.

$$T = 518.688 \times (1 - 6.875\times10^{-6}H)$$

And the relationship between density of air and vertical displacement is

$$\rho/\rho_o = (P/RT)(P_o/RT_o)$$
$$= (P/P_o)(T_o/T)$$

Since the effects of changing altitude become larger, it is necessary to modify the equation.

Let $K_9 = (\rho/\rho_o)^{1.2}$

Then K_1 is proportional to the K_9 and K_2 is inverse proportional to the K_9

Thus in this case

$$\rho/\rho_o = (P/P_o)(T_o/T)$$
$$= (1 - 6.875\times10^{-6}\times4{,}000)^{5.2561} (1 - 6.875\times10^{-6} \times 4{,}000)$$
$$= 0.88809$$

$$K_1 = K_{1_{ssl}} \times (0.88809)^{1.2}$$
$$= .0139976$$

$$K_2 = K_{2_{ssl}} \; (0.88809)^{1.2}$$
$$= 2.306\times10^8$$

$$\text{Thrust required} = 0.0139976V^2 + 2.306\times10^8 \, V^2$$

The military thrust will be reduced as discussed in Chapter 2

$$T_{avail} = T_{ssl} \, \delta \, \sigma^{(1.2)}$$

$$\delta = P_{4{,}000ft} / P_{ssl}$$
$$= 1{,}827.69/2{,}116.22$$
$$= 0.8636578$$

$$\sigma = \rho_{4{,}000ft} / \rho_{ssl}$$
$$= 0.0021109/0.0023769$$
$$= [illegible]$$

[illegible]

Fuel flow;

$$F_1 = 26{,}000 - 0.64 \times \text{Altitude}$$
$$= 26{,}000 - 0.64 \times 4{,}000$$
$$= 23{,}440$$
$$F_2 = F_1 - 16.65 \times \text{Drag Index}$$
$$= 23{,}440 - 16.65 \times 20$$
$$= 23{,}107$$
$$F_3 = F_2 - 0.002 \times (\text{Gross weight} - 40{,}000)$$
$$= 23{,}107 - 0.002 \times (40{,}000 - 40{,}000)$$
$$= 0.86463$$

$$C = 19{,}979\ 23{,}107$$
$$= 0.86463$$

Fuel flow = Thrust required/0.86463

Table 16 and Figure 3.5 show the results of calculation and deviation from hand book data.

E. VARIATION NUMBER 4 - WEIGHT, DRAG INDEX AND ALTITUDE

Given condition

Gross weight	50,000 lbs
Drag Index	40
Altitude	8,000 ft

If all conditions are changed from base line conditions, it is neceassary to consider for each changed condition step by step, same as previous sections.

At first, think about gross weight change. The changing gross weight affects the k_2 only, because K_1 is independent of gross weight.

$$K_2 = 2W^2\ (\rho S \pi e AR)$$

As you can see in section 3.2

$$K_{2_{50{,}000lbs}} = K_{2_{40{,}000lbs}} (50{,}000\ 40{,}000)^{2.7}$$
$$= 2.0\times10^8 (1.25)^{2.7}$$
$$= 3.653\times10^8$$

TABLE 16

THRUST REQUIRED AND FUEL FLOW WITH ALTITUDE VARIATION

ALTITUDE	4,000 FT	GROSS WEIGHT	40,000 LBS		DRAG INDEX	20
MILITARY	THRUST	19,979 LB		THRUST	FUEL FLOW x 0.8582	
	FUEL FLOW	23,280 LBS/H				
AIR SPEED (KTAS)	FUEL FLOW (LBS/H)		THRUST REQUIRED (LB)		DEVIATION (%)	
	HAND BOOK	CALCULATION	HAND BOOK	CALCULATION	THRUST	FUEL FLOW
360	6,818	6,632	5,851	5,798	-0.9	-2.7
400	7,931	7,886	6,806	6,893	1.3	-0.6
440	9,219	9,320	7,912	8,147	3.0	1.1
480	10,716	10,925	9,196	9,550	3.8	1.9
520	12,686	12,692	10,887	11,095	1.9	0.1
560	15,171	14,797	13,020	12,936	-0.6	-2.5
600	18,641	18,412	15,998	16,095	0.6	-1.2
MAXIMUM ENDURANCE		FUEL FLOW	5,390			(LBS/H)
		THRUST (MINIMUM)	5,390x0.8582 = 4,625			(LB)
		AIR SPEED	267			(KTAS)

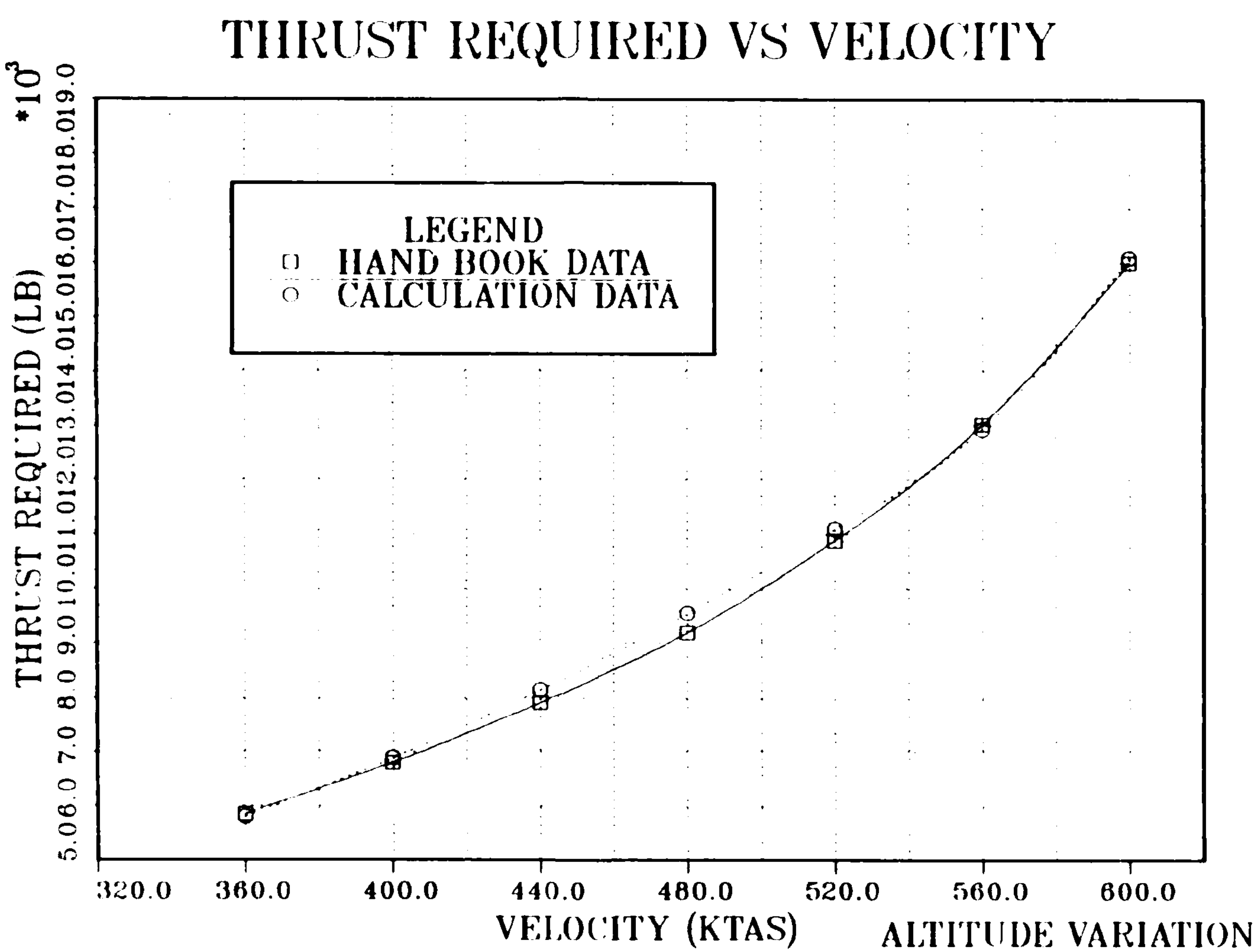

Figure 3.5a Thrust Required with Altitude variation.

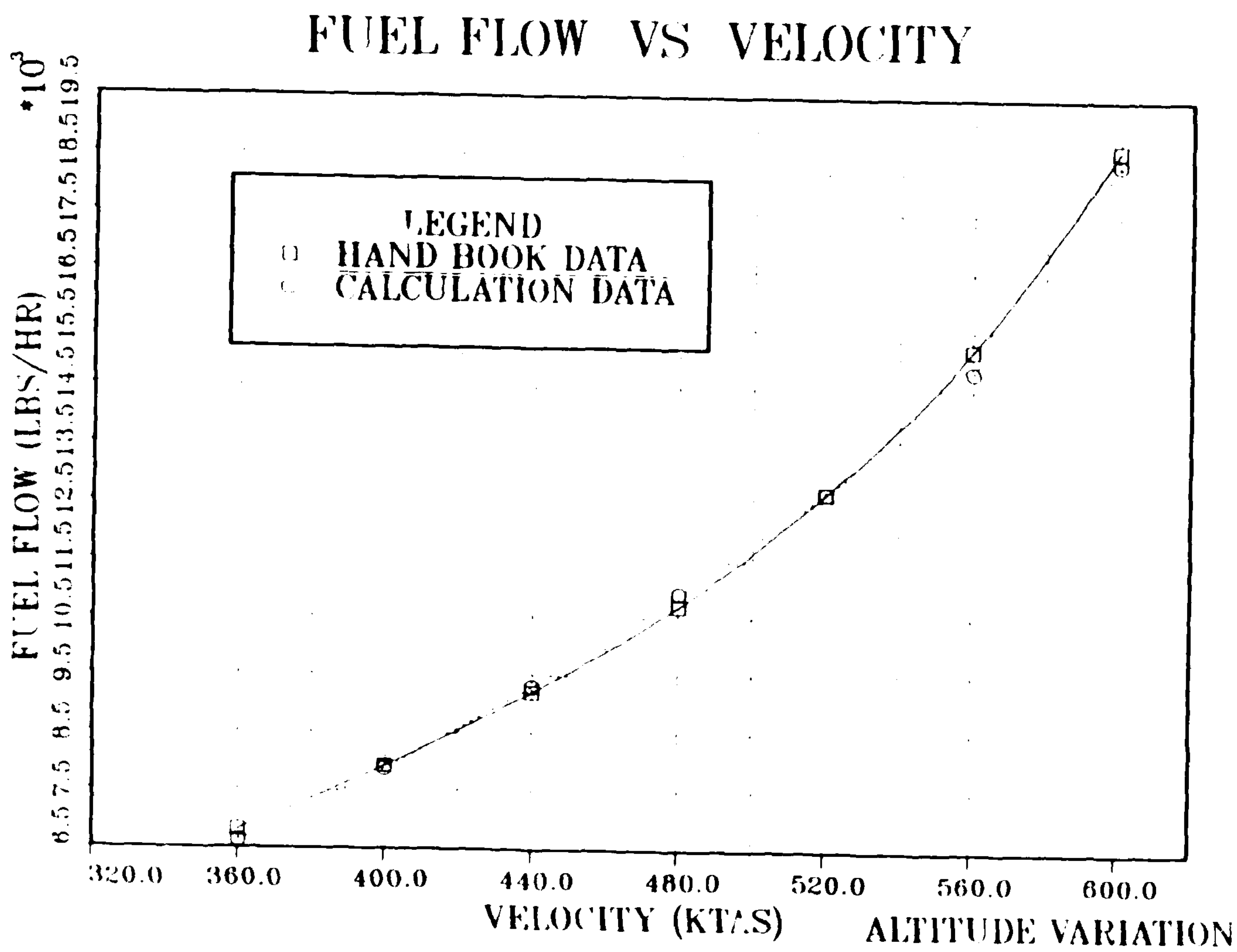

Figure 3.5b Fuel Flow with Altitude variation.

Next, think about the drag index change. If the drag index is changed, it is necessary to consider K_1 only, because the K_2 is independent of the drag index change.

K_1 equation, function of drag index is

$$K_1 = 0.0135 + 0.000132 \times \text{Drag Index}$$

Thus

$$K_{1_{D140}} = 0.0135 + 0.000132 \times 40$$

$$= 0.01878$$

And finally, think about the altitude change.

As you can see in section 3.4, the altitude change affects both K_1 and K_2, because K_1 and K_2 are function of air density. The equation in section 3.4 can be used in this step.

$$\rho/\rho_o = (P/RT)(P_o/RT_o)$$

$$= (P/P_o)(T_o/T)$$

Let $K_9 = (\rho/\rho_o)^{1.2}$

$K_1 = f(\rho)$

$K_2 = f(1/\rho)$

Thus

$$K_1 = K_{1_{D140}} \times (\rho_{8,000ft}/\rho_{ssl})^{1.2}$$

$$= 0.01878 \times (0.0018683/0.0023769)^{1.2}$$

$$= 0.01406755$$

$$K_2 = K_{2_{50,000lbs}} / (\rho_{8,000ft}/\rho_{ssl})^{1.2}$$

$$= 3.653 \times 10^8 / (0.0018683/0.0023769)^{1.2}$$

$$= 2.9157 \times 10^8$$

Thrust required $= 0.01407V^2 + 2.9157 \times 10^8\ V^2$

The military thrust will be computed as discussed in Chapter 2

$$T_{8,000ft} = T_{ssl}(\delta\ \sigma^{1/2})$$

$$\delta = P_{8,000ft}/P_{ssl}$$

$$= 1571.88/2116.22$$

$$= 0.742777$$

$$\sigma = \rho_{8,000ft} / \rho_{ssl}$$

$$= 0.0018683/0.0023769$$

$$= 0.786024$$

$$T_{8,000ft} = 21,800 \text{ x } (0.742777/0.786024^{1/2})$$

$$= 18,264 \text{lb}$$

Fuel flow;

$$F_1 = 26,000 - 0.64 \text{ x Altitude}$$

$$= 26,000 - 0.64\text{x}8,000$$

$$= 20,880$$

$$F_2 = (F_1 + 333.3) - 22.1 \text{ x Drag Index}$$

$$= 21,213.3 - 22.1 \text{ x } 40$$

$$= 20,329.3$$

$$F_3 = F_2 - 0.002 \text{ x (Gross weight - 40,000)}$$

$$= 20,329.3 - 0.002 \text{ x } (50,000 - 40,000)$$

$$= 20,309.3$$

$$C = \text{Military thrust / Fuel flow}$$

$$= 18,264/20,309.3$$

$$= 0.8993$$

$$\text{Fuel flow} = \text{Thrust required}/0.8993$$

Table 17 and Figure 3.6 show the results of calculation and deviation from hand book data.

F. VARIATION NUMBER 5 - WEIGHT, DRAG INDEX AND ALTITUDE

We have seen how the fuel flow was changed with various conditions. Not much time is needed to find the fuel flow for conditions which can be read directly from hand book, but most of the conditions do not correspond to the published conditions which can be read directly from the hand book data sheets. Thus to find the fuel flow for specific conditions that are not printed in the hand book, one must spend lots of time to interpolate each item. The specific fuel flow will now be computed by both methods.

TABLE 17

THRUST REQUIRED AND FUEL FLOW WITH ALL CONDITIONS VARIATION

ALTITUDE	8,000 FT	GROSS WEIGHT	50,000 LBS		DRAG INDEX	40
MILITARY	THRUST	18,264 LB	THRUST		FUEL FLOW x 0.90505	
	FUEL FLOW	20,180 LBS/H				
AIR SPEED (KTAS)	FUEL FLOW (LBS/H)		THRUST REQUIRED (LB)		DEVIATION (%)	
	HAND BOOK	CALCULATION	HAND BOOK	CALCULATION	THRUST	FUEL FLOW
360	7,274	7,119	6,583	6,520	-1.0	-2.1
400	8,135	8,177	7,363	7,489	1.7	0.5
440	9,239	9,447	8,362	8,652	3.5	2.3
480	10,668	10,905	9,655	9,988	3.4	2.2
520	12,512	12,539	11,324	11,483	1.4	0.2
560	15,126	14,639	13,690	13,407	-2.1	-3.2
600						
MAXIMUM ENDURANCE		FUEL FLOW	6,600			(LBS/H)
		THRUST (MINIMUM)	6,600x0.90505 = 5,793			(LB)
		AIR SPEED	318			(KTAS)

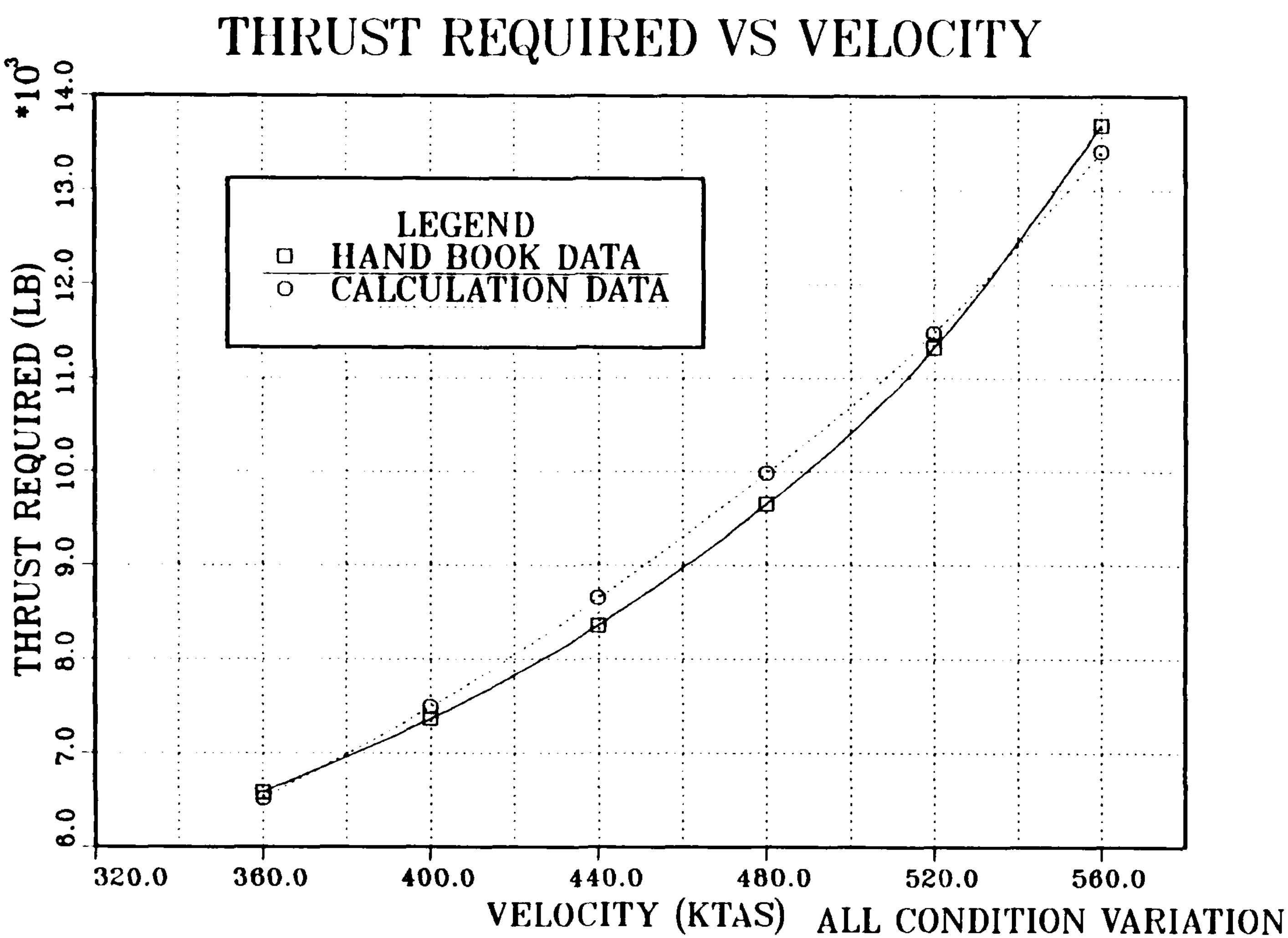

Figure 3.6a Thrust Required with All Condition variation.

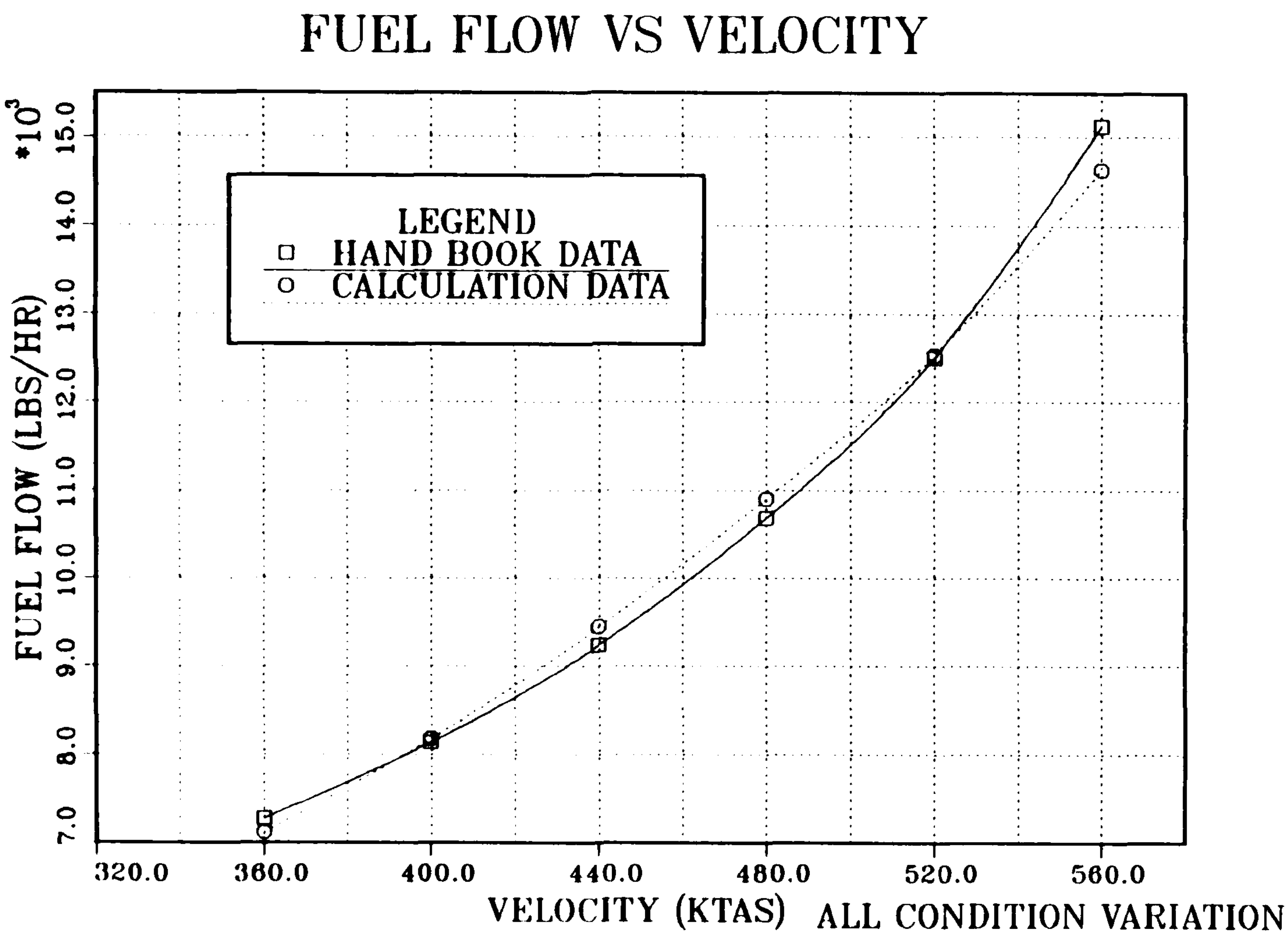

Figure 3.6b Fuel Flow with All Condition variation.

Given condition

Gross weight	48,000 lbs
Drag Index	27
Altitude	11,000 ft
Velocity	455 KTS

The procedure to compute the K_1 and K_2 is the same as in the previous section. Consider each different condition.

1. **Gross weight**

$$K_{2_{48,000lbs}} = K_{2_{40,000lbs}} \times (48,000/40,000)^{2.7}$$
$$= 2.0\times10^8 \times (1.2)^{2.7}$$
$$= 3.27\times10^8$$

K_1 is independent of changing gross weight.

2. **Drag index**

$$K_1 = 0.0135 + 0.000132 \times \text{Drag Index}$$
$$= 0.0135 + 0.000132 \times 27$$
$$= 0.01706$$

K_2 is independent of changing Drag Index.

3. **Altitude**

$$K_1 = K_{1_{DI27}} \times (\rho_{11,000ft} / \rho_{ssl})^{1.2}$$
$$= 0.01706 \times (0.0017007/0.0023769)^{1.2}$$
$$= 0.01142$$

$$K_2 = K_{2_{48,000lb}} / (\rho_{11,000ft} / \rho_{ssl})^{1.2}$$
$$= 3.27\times10^8/(0.0017708/0.0023769)^{1.2}$$
$$= 4.8893\times10^8$$

$$\text{Thrust required} = 0.01142V^2 + 4.8893\times10^8/V^2$$
$$= 0.01142 \times (455\times1.69)^2 + 4.8893\times10^8/(455\times1.69)^2$$
$$= 7,572 \text{ lb}$$

Compute the factor C:

Military thrust = 21,800 - 0.4 x Altitude

= 21,800 - 0.4x11,000

= 17,400 lb

Fuel flow;

F_1 = 20,840 - 0.37 x (Altitude - 8,000)

= 20,840 - 0.37 x (11,000 - 8,000)

= 19,730

F_2 = (F_1 + 333.3) - 22.1 x Drag Index

= 20,063.3 - 22.1 x 27

= 19,466.6

F_3 = F_2 - 0.002 x (Gross weight - 40,000)

= 19,466.6 - 0.002 x (48,000 - 40,000)

= 19,450.6

C = Military thrust Fuel flow

= 17,400 19450.6

= 0.89457

Thus

Fuel flow = Thrust required C

= 7,572 0.89457

= 8,464 lbs hr

Now the fuel flow for given conditions will be computed by interpolating the hand book data. Figure 3.7 shows the whole diagram to compute the desired fuel flow of given conditions.

Computations for the steps are:

1) F1 = (8873 -7988)x(27 - 20) (40 - 20) + 7988 = 8297.75
2) F2 = (10324 - 9223)x(27 - 20) (40 - 20) + 9223 = 9608.35
3) F3 = (455 - 440)x(9608.35 - 8297.75) (480 - 440) + 8297.75 = 8789.23
4) G1 = (7729 - 6962)x(27 - 20) (40 - 20) + 6962 = 7230.45
5) G2 = (8840 - 7933)x(27 -20) (40 -20) + 7933 = 8250.45
6) G3 = (455 - 440)x(8250.45 - 7230.45) (480 - 440) + 7230.75 = 7613.25
7) I4 = (11000 - 8000)x(7613.25 - 8789.23) (12000 - 8000) + 8789.23 = 7907.24
8) H1 = (9239 - 8366)x(27 - 20) (40 - 20) + 8366 = 8671.55

DIAGRAM FOR DESIRED FUEL FLOW

GROSS WEIGHT 40,000 LBS								
ALTITUDE 8,000 FT				11,000 FT	ALTITUDE 12,000 FT			
A/S	DRAG INDEX				A/S	DRAG INDEX		
KTAS	20	27	40		KTAS	20	27	40
440	7988	F_1	8873		440	6962	G_1	7729
455		F_3		L_4	455		G_3	
480	9223	F_2	10324		480	7933	G_2	8840

GROSS WEIGHT 50,000 LBS								
ALTITUDE 8,000 FT				11,000 FT	ALTITUDE 12,000 FT			
A/S	DRAG INDEX				A/S	DRAG INDEX		
KTAS	20	27	40		KTAS	20	27	40
440	8366	H_1	9239		440	7447	J_1	8192
455		H_3		M_4	455		J_3	
480	9522	H_2	10688		480	8315	J_2	9216

G/WT 40,000 LBS		48,000 LBS	G/WT 50,000 LBS	
A/S	DRAG INDEX		A/S	DRAG INDEX
KTAS	27		KTAS	27
455	L_4	N_4	455	M_4

Figure 3.7 Diagram for Desired Fuel Flow.

[illegible] H2 = [illegible] 22 [illegible] 20 [illegible] 40 [illegible] 20 [illegible] 22 [illegible]

H3 = 4[illegible] 440 [illegible]

11 T1 = 8142 [illegible] 7447 x (27 [illegible] 20 [illegible] 40 [illegible] 20 [illegible] 7447 [illegible]

12 T2 = 9216 [illegible] 8315 x (27 [illegible] 20 [illegible] 40 [illegible] 20 [illegible] = 8630.35

13 T3 = (455 [illegible] 440) x (8630.35 [illegible] 7767.78 [illegible] (480 [illegible] 440) + 7767.78 = [illegible]

14 M4 = (11000 [illegible] 8000) x (8053.73 [illegible] 4143.51 [illegible] (12000 [illegible] 8000) + [illegible] = 8326.174

15 N4 = (48000 [illegible] 40000) x (8326.17 [illegible] 7907.24) / (50000 [illegible] 40000) + 7907.24 = 8242.39

Desired fuel flow for given condition is 8,242 lbs/hr

Thus deviation of the results

(8,464 - 8,242)/8,242 x 100 = 2.7 %

IV. CONCLUSION

[illegible]

[illegible] determine the fuel flow [illegible] flight manual, [illegible] be necessary [illegible] these steps [illegible] very time consuming and the probability of making a mistake [illegible] calculating so many interpolations.

Thus the computer program to compute the fuel flow rate can [illegible] save time for many conditions which must be changed continuously during flight [illegible] reduces the probability of making a mistake in interpolations steps.

It is recommended that a similar computer program be developed for specific flight stages, i.e., take-off, climb, or landing, using the same concepts that were used in this program.

APPENDIX A

COMPUTER PROGRAM (BASIC)

```
00010 K2=2.0E+08 ' REM ** BASED ON S.S.L. AND GROSS WEIGHT 40000LBS **
00020 PRINT 'INPUT ALTITUDE,GROSS WEIGHT,DRAG INDEX,AIR SPEED,A/S INC'
00030 INPUT A,W,I9,V1,VP
00040 K1=(0.0135+.000132*I9) ' REM ** K1 VARIES WITH DRAG INDEX **
00050 D4=1-6.875E-6*A ' REM ** ICAO SCALE BASE **
00060 T1=518.688*D4 ' REM ** ABSOLUTE TEMP (R) BASED ON ALTITUDE **
00070 D5=D4**5.2561 ' REM ** PRESSURE RATIO BASED ON ALTITUDE **
00080 A9=(1.4*1714.8*T1) ' REM ** SPEED OF SOUND (SQUARED) **
00090 K9=(D5/D4)**1.2 ' REM ** ALTITUDE CORRECTION FOR 'Q' **
00100 K1=K1*K9 ' REM ** PA*M**2 ALTITUDE CORRECTION **
00110 K2=K2/K9 ' REM ** PA*M**2 ALTITUDE CORRECTION **
00120 K2=K2*(W/40000)**2.7 'REM** WEIGHT OTHER THAN 40000LB CORRECTION**
00160 PRINT 'K1=',K1,'K2=',K2
00170 PRINT TAB(1),'VEL(KTS)',TAB(12);'MACH';TAB(20);'PAR DRAG';
00180 PRINT TAB(34),'PRO DRAG',TAB(48),'MACH DRAG';TAB(62);'FUEL FLOW'
00190 V2=V1*1.68894
00200 M=V2/A9**0.5
00210 D8=1 ' REM ** CDO MACH CORRECTION BASE **
00220 IF  M>.8  THEN 480
00230 D1=K1*V2**2 ' REM ** PARASITE DRAG (SUBSONIC) **
00240 D3=D1*D8 ' REM ** PARASITE DRAG (MACH CDO) **
00250 D2=K2/V2**2 ' REM ** INDUCED DRAG **
00260 D0=D3+D2 ' REM ** MACH DRAG **
00270 D7=D1+D2 ' REM ** SUBSONIC TOTAL DRAG **
00280 T=21800-A*0.4 ' REM ** MIL THRUST OTHER THAN S.S.L. CORRECTION **
00290 IF A<=8000 THEN 320
00300 F1=20840-(A-8000)*0.37! REM ** F/FLOW CORRECTTION ABOVE 8000 FT**
00310 GO TO 330
00320 F1=26000-0.64000*A ! REM ** F/FLOW CORRECTION BELOW 8000 FT **
00330 IF I9>20 THEN 360
00340 F2=F1-16.65*I9 ! REM ** F/FLOW CORRECTION BELOW DRAG INDEX 20 **
00350 GO TO 370
00360 F2=(F1+333.3)-22.10*I9 ! * F/FLOW CORRECTION ABOVE DRAG INDEX 20 *
00370 IF W<=40000THEN 400
00380 F3=F2-(W-40000)*0.002 ! REM ** F/FLOW CORRECTION WITH WEIGHT **
00390 GO TO 410
00400 F3=F2
00410 C=T/F3 ! REM ** COEFFICIENT FOR FUEL FLOW **
00420 FF=D0/C ! REM ** FUEL FLOW **
00430 PRINT TAB(3);V1;TAB(8);M;TAB(18);D2;TAB(32);D3;TAB(46);D0;
00440 PRINT TAB(60);FF
00450 V1=V1+VP
00460 IF  V1>610  THEN 510
00470 GOTO 190
00480 D9=.55*(SIN((M-.8)/.8)**3)/.0180
00490 D8=1+D9
00500 GOTO 230
00510 PRINT 'C=';C,'T=';T
00520 STOP
00530 END
```

APPENDIX B

COMPUTER PROGRAM (HAND-HELD CALCULATOR)

```
01♦LBL "PP"
02 1
03 STO 24
04 RCL 00
05 "ALT=?"
06 PROMPT
07 STO 00
08 RCL 01
09 "W=?"
10 PROMPT
11 STO 01
12 RCL 02
13 "DI=?"
14 PROMPT
15 STO 02
16♦LBL "VV"
17 RCL 03
18 "V(KTS)=?"
19 PROMPT
20 STO 03
21 RCL 02
22 .000132
23 *
24 .0135
25 +
26 STO 04
27 6.875 E-06
28 RCL 00
29 *
30 CHS
31 1
32 +
33 STO 05
34 518.68
35 *
36 STO 06
37 1716.5
38 *
39 1.4
40 *
41 STO 07
42 RCL 05
43 ENTER↑
44 5.2561
45 Y↑X
46 STO 08
47 1116
48 X↑2
49 RCL 07
50 /
51 RCL 08
52 *
53 STO 09
54 RCL 04
55 RCL 03
56 /
57 STO 04
58 2.0 E03
59 RCL 09
60 /
61 STO 10
62 RCL 01
63 40000
64 /
65 2.7
66 Y↑X
67 RCL 10
68 *
69 STO 10
70 RCL 03
71 1.68894
72 *
73 STO 11
74 RCL 07
75 SQRT
76 1/X
77 RCL 11
78 *
79 STO 12
80 .8
81 X<=Y?
82 GTO "MM"
83♦LBL "RR"
84 RCL 11
85 X↑2
86 RCL 04
87 *
88 STO 13
89 RCL 24
90 *
91 STO 14
92 RCL 11
93 X↑2
94 1/X
95 RCL 10
96 *
97 STO 15
98 RCL 14
99 +
100 STO 16
101 RCL 15
102 RCL 17
103 +
104 STO 17
105 RCL 00
106 .4
107 *
108 CHS
109 21800
110 +
111 STO 18
112 8000
113 RCL 00
114 X<=Y?
115 GTO "PP"
116 RCL 00
117 8000
118 -
119 .37
120 *
121 CHS
122 20840
123 +
124 STO 19
125 GTO "LL"
126♦LBL "PP"
127 RCL 00
128 .64
129 *
130 CHS
131 26000
132 +
133 STO 19
134♦LBL "LL"
135 20
136 RCL 02
137 X>Y?
138 GTO "DD"
139 16.65
140 *
141 CHS
142 RCL 19
143 +
144 STO 20
145 GTO "EE"
146♦LBL "DD"
147 22.1
148 *
149 CHS
150 333.3
151 +
152 RCL 19
153 +
154 STO 20
155♦LBL "EE"
156 RCL 01
157 40000
158 -
159 .002
160 *
161 CHS
162 RCL 20
163 +
164 STO 21
165 1 X
166 RCL 18
167 *
168 STO 22
169 1/X
170 RCL 16
171 *
172 STO 23
173 RCL 03
174 FIX 0
175 "V(KTS)=
176 ARCL X
177 AVIEW
178 STOP
179 RCL 12
180 FIX 3
181 "MACH="
182 ARCL X
183 AVIEW
184 STOP
185 RCL 15
186 FIX 1
187 "D(I)="
188 ARCL X
189 AVIEW
190 STOP
191 RCL 14
192 "D(P)="
193 ARCL X
194 AVIEW
195 STOP
196 RCL 16
197 "D(M)="
198 ARCL X
199 AVIEW
200 STOP
201 RCL 23
202 "FF="
203 ARCL X
204 AVIEW
205 STOP
206 "NEW V? 1,0 "
207 PROMPT
208 X=0?
209 STOP
210 GTO "VV"
211♦LBL "MM"
212 RCL 12
213 .8
214 -
215 .8
216 /
217 ENTER↑
218 3
219 Y↑X
220 SIN
221 .55
222 *
223 .018
224 /
225 1
226 +
227 STO 24
228 GTO "PP"
229 .END.
```

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